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NATIONAL ADVISORY COMMITTEE  
FOR AERONAUTICS

DATA TR. 903

REPORT No. 903

THEORETICAL AND EXPERIMENTAL DATA FOR A  
NUMBER OF NACA 6A-SERIES AIRFOIL SECTIONS

By LAURENCE E. LOFTIN, Jr.

1948

## AERONAUTIC SYMBOLS

### I. FUNDAMENTAL AND DERIVED UNITS

	Symbol	Metric		English	
		Unit	Abbreviation	Unit	Abbreviation
Length	$l$	meter	m	foot (or mile)	ft (or mi)
Time	$t$	second	s	second (or hour)	sec (or hr)
Force	$F$	weight of 1 kilogram	kg	weight of 1 pound	lb
Power	$P$	horsepower (metric)		horsepower	hp
Speed	$v$	(kilometers per hour) (meters per second)	kph mps	miles per hour feet per second	mph fps

### II. GENERAL SYMBOLS

$w$	Weight = $mg$	Kinematic viscosity
$g$	Standard acceleration of gravity = $9.80665 \text{ m/s}^2$ or $32.1740 \text{ ft/sec}^2$	Density (mass per unit volume)
$m$	Mass = $\frac{W}{g}$	Standard density of dry air, $0.12497 \text{ kg-m}^{-3} \cdot \text{s}^3$ at $15^\circ \text{C}$ and 760 mm, or $0.002378 \text{ lb-ft}^{-4} \text{ sec}^3$
$I$	Moment of inertia = $mL^2$ . (Indicate axis of radius of gyration $k$ by proper subscript.)	Specific weight of "standard" air, $1.2255 \text{ kg/m}^3$ or $0.07651 \text{ lb/cu ft}$
$\mu$	Coefficient of viscosity	

### III. AERODYNAMIC SYMBOLS

$S$	Area	Angle of setting of wings (relative to thrust line)
$S_w$	Area of wing	Angle of stabilizer setting (relative to thrust line)
$G$	Gap	Resultant moment
$\delta$	Span	Resultant angular velocity
$c$	Chord	$R = \frac{Vl}{\mu}$ where $l$ is a linear dimension (e.g., for an airfoil of 1.0 ft chord, 100 mph, standard pressure at $15^\circ \text{C}$ , the corresponding Reynolds number is 935,400; or for an airfoil of 1.0 m chord, 100 mps, the corresponding Reynolds number is 6,865,000)
$A$	Aspect ratio	Angle of attack
$V$	True air speed	Angle of downwash
$q$	Dynamic pressure, $\frac{1}{2} \rho V^2$	Angle of attack, infinite aspect ratio
$L$	Lift, absolute coefficient $C_L = \frac{L}{qS}$	Angle of attack, induced
$D$	Drag, absolute coefficient $C_D = \frac{D}{qS}$	Angle of attack, absolute (measured from zero-lift position)
$D_0$	Profile drag, absolute coefficient $C_{D_0} = \frac{D_0}{qS}$	Flight-path angle
$D_i$	Induced drag, absolute coefficient $C_{D_i} = \frac{D_i}{qS}$	
$D_p$	Parasite drag, absolute coefficient $C_{D_p} = \frac{D_p}{qS}$	
$C$	Cross-wind force, absolute coefficient $C_C = \frac{C}{qS}$	

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By LAURENCE K. LOFTIN, Jr.  
Langley Memorial Aeronautical Laboratory  
Langley Field, Va.

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## REPORT No. 903

### THEORETICAL AND EXPERIMENTAL DATA FOR A NUMBER OF NACA 6A-SERIES AIRFOIL SECTIONS

By LAURENCE K. LOFTIN, Jr.

#### SUMMARY

The NACA 6A-series airfoil sections were designed to eliminate the trailing-edge cusp which is characteristic of the NACA 6-series sections. Theoretical data are presented for NACA 6A-series basic thickness forms having the position of minimum pressure at 30, 40, and 50 percent chord and with thickness ratios varying from 6 percent to 15 percent. Also presented are data for a mean line designed to maintain straight sides on the cambered sections.

The experimental results of a two-dimensional wind-tunnel investigation of the aerodynamic characteristics of five NACA 64A-series airfoil sections and two NACA 63A-series airfoil sections are presented. An analysis of these results, which were obtained at Reynolds numbers of  $3 \times 10^6$ ,  $6 \times 10^6$ , and  $9 \times 10^6$ , indicates that the section minimum-drag and maximum-lift characteristics of comparable NACA 6-series and 6A-series airfoil sections are essentially the same. The quarter-chord pitching-moment coefficients and angles of zero lift of NACA 6A-series airfoil sections are slightly more negative than those of corresponding NACA 6-series airfoil sections. The position of the aerodynamic center and the lift-curve slope of smooth NACA 6A-series airfoil sections appear to be essentially independent of airfoil thickness ratio in contrast to the trends shown by NACA 6-series sections. The addition of standard leading-edge roughness causes the lift-curve slope of the newer sections to decrease with increasing airfoil thickness ratio.

#### INTRODUCTION

Much interest is being shown in airfoil sections having small thickness ratios because of their high critical Mach numbers. The NACA 6-series airfoil sections of small thickness have relatively high critical Mach numbers but have the disadvantage of being very thin near the trailing edge, particularly when the sections considered have the position of minimum pressure well forward on the basic thickness form. The thin trailing-edge portions lead to difficulties in structural design and fabrication. In order to overcome these difficulties, the trailing-edge cusp has been removed from a number of NACA 6-series basic thickness forms and the sides of the airfoil sections made straight from approximately 80 percent chord to the trailing edge. These new sections are designated NACA 6A-series airfoil sections. A special mean line, designated the  $a=0.8$  (modified) mean

line, has also been designed to maintain straight sides on the cambered sections.

This paper presents theoretical pressure-distribution data and ordinates for NACA 6A-series basic thickness form covering a range of thickness ratios extending from 6 to 15 percent and a range of positions of minimum pressure extending from 30 percent to 50 percent chord.

The aerodynamic characteristics of seven NACA 6A-series airfoil sections as determined in the Langley two-dimensional low-turbulence pressure tunnel are also presented. These data are analyzed and compared with similar data for NACA 6-series airfoil sections of comparable thickness and design lift coefficient.

#### COEFFICIENTS AND SYMBOLS

$c_d$	section drag coefficient
$c_{d_{min}}$	minimum section drag coefficient
$c_l$	section lift coefficient
$c_{l_d}$	design section lift coefficient
$c_{l_{max}}$	maximum section lift coefficient
$c_{m_{ac}}$	section pitching-moment coefficient about aerodynamic center
$c_{m_{c/4}}$	section pitching-moment coefficient about quarter-chord point
$\alpha_0$	section angle of attack
$\alpha_i$	section angle of attack corresponding to design lift coefficient
$\frac{dc_l}{d\alpha_0}$	section lift-curve slope
$V$	free-stream velocity
$v$	local velocity
$\Delta v$	increment of local velocity
$\Delta v_a$	increment of local velocity caused by additional type of load distribution
$P_R$	resultant pressure coefficient; difference between local upper-surface and lower-surface pressure coefficient
$R$	Reynolds number
$c$	airfoil chord length
$x$	distance along chord from leading edge
$y$	distance perpendicular to chord
$y_c$	mean-line ordinate
$a$	mean-line designation; fraction of chord from leading edge over which design load is uniform
$\psi$	airfoil design parameter (reference 1)

### THEORETICAL CHARACTERISTICS OF AIRFOILS

**Designation.**—The system used for designating the new airfoil sections is the same as that employed for the NACA 6-series sections (reference 1) except that the capital letter "A" is substituted for the dash which appears between the digit denoting the position of minimum pressure and that denoting the ideal lift coefficient. For example, the NACA 64<sub>1</sub>-212 becomes the NACA 64<sub>1</sub>A212 when the cusp is removed from the trailing edge. In the absence of any further modification of the designation, the cambered airfoils are to be considered as having the  $a=0.8$  (modified) mean line.

**Basic thickness forms.**—The theoretical methods by which the basic thickness forms of the NACA 6-series family of airfoil sections were derived in order to have pressure distributions of a specified type are described in reference 1. Removing the trailing-edge cusp was accomplished by increasing the value of the airfoil design parameter  $\psi$  (reference 1) corresponding to the rear portion of the airfoil until the airfoil ordinates formed a straight line from approximately 80 percent chord to the trailing edge. Once the final form of the  $\psi$  curves was established, the new pressure distribu-

tions corresponding to the modified thickness forms were calculated by the usual methods as described in reference 1.

A comparison of the theoretical pressure distributions of an NACA 64<sub>1</sub>-012 airfoil section and an NACA 64<sub>1</sub>A012 airfoil section (fig. 1) indicates that removing the trailing-edge cusp has little effect upon the velocities around the section. A slight reduction of the peak negative pressure and flatter pressure gradient over the forward and rearward portions of the airfoil section seem to be the principal effects. The theoretical calculations also indicate the presence of a trailing-edge stagnation point caused by the finite trailing-edge angle of the NACA 6A-series sections. This stagnation point is, of course, never realized experimentally.

Ordinates and theoretical pressure-distribution data for NACA 6A-series basic thickness forms having the position of minimum pressure at 30, 40, and 50 percent chord are presented in figure 2 for airfoil thickness ratios of 6, 8, 10, 12, and 15 percent. If intermediate thickness ratios involving a change in thickness of not more than 1 to 2 percent are desired, the ordinates of the basic thickness forms may be scaled linearly without seriously altering the gradients of the theoretical pressure distribution.

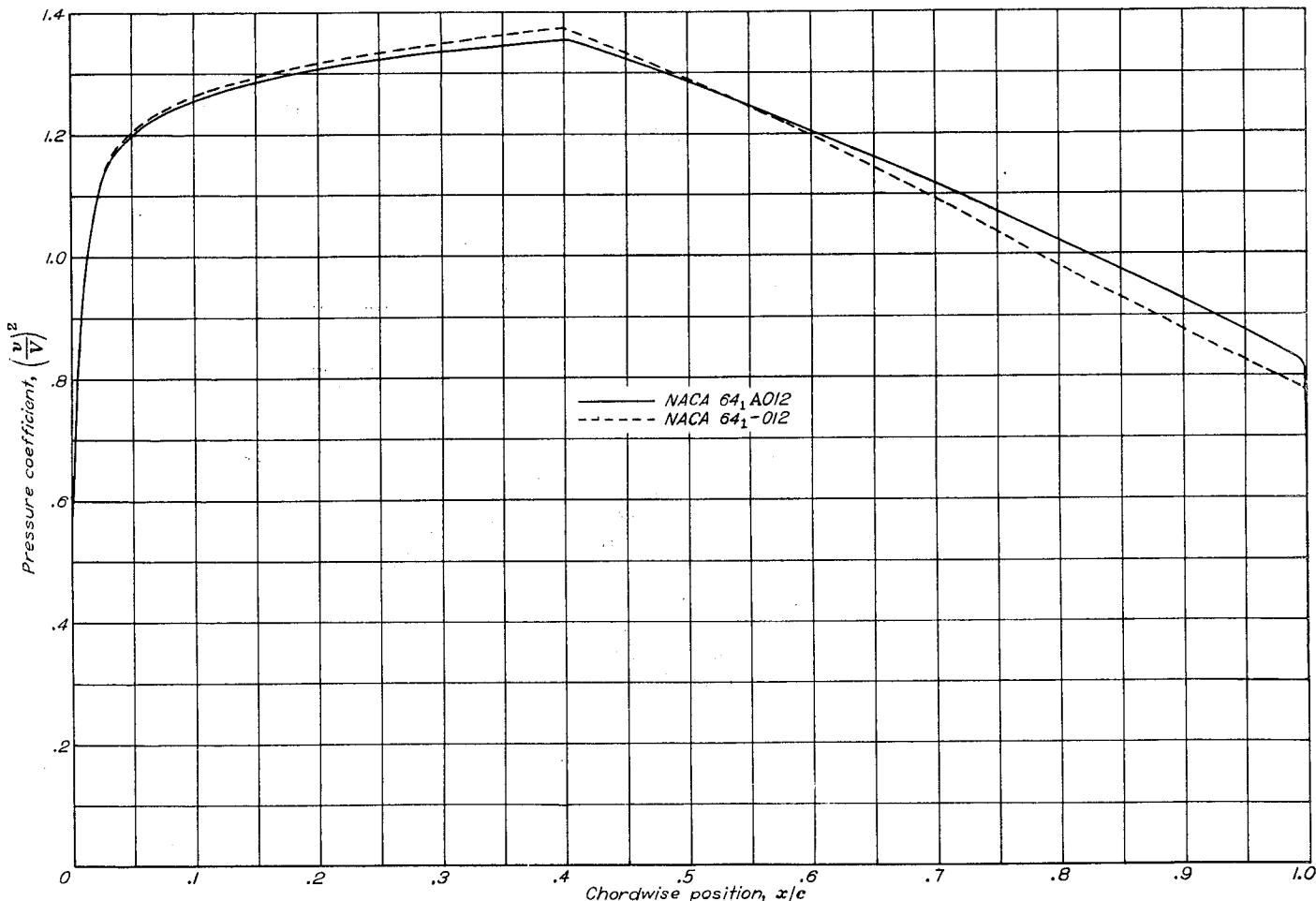


FIGURE 1.—Comparison of theoretical pressure distribution at zero lift of the NACA 64<sub>1</sub>-012 and the NACA 64<sub>1</sub>A012 airfoil sections.

NACA 63A006 BASIC THICKNESS FORM

	$x$ (percent c)	$y$ (percent c)	$(v/V)^2$	$v/V$	$\Delta v_a/V$
	0	0	0	0	0
	.5	.495	.900	.949	4.560
	.75	.505	1.063	1.031	2.079
	1.25	.754	1.086	1.042	1.794
	2.5	1.045	1.112	1.055	1.076
	5.0	1.447	1.134	1.065	.693
	7.5	1.747	1.142	1.068	.563
	10	1.969	1.150	1.072	.485
	15	2.362	1.159	1.077	.383
	20	2.631	1.165	1.079	.321
	25	2.820	1.168	1.081	.278
	30	2.942	1.170	1.082	.244
	35	2.986	1.169	1.081	.217
	40	2.985	1.162	1.078	.195
	45	2.914	1.151	1.073	.175
	50	2.788	1.138	1.067	.158
	55	2.613	1.120	1.058	.140
	60	2.396	1.100	1.049	.126
	65	2.153	1.079	1.039	.112
	70	1.859	1.057	1.028	.098
	75	1.556	1.035	1.017	.085
	80	1.248	1.010	1.005	.072
	85	.939	.986	.983	.060
	90	.630	.964	.982	.047
	95	.322	.939	.969	.033
	100	.013	0	0	0

L. E. radius: 0.265 percent c

T. E. radius: 0.014 percent c

NACA 63A008 BASIC THICKNESS FORM

	$x$ (percent c)	$y$ (percent c)	$(v/V)^2$	$v/V$	$\Delta v_a/V$
	0	0	0	0	0
	.5	.658	.850	.922	3.466
	.75	.791	1.034	1.017	1.961
	1.25	1.003	1.080	1.039	1.674
	2.5	1.391	1.132	1.064	1.344
	5.0	1.930	1.168	1.081	.967
	7.5	2.332	1.185	1.089	.889
	10	2.656	1.198	1.095	.762
	15	3.155	1.212	1.101	.484
	20	3.515	1.221	1.105	.383
	25	3.766	1.227	1.108	.322
	30	3.926	1.230	1.109	.279
	35	3.985	1.228	1.108	.246
	40	3.978	1.219	1.104	.218
	45	3.878	1.204	1.097	.195
	50	3.705	1.189	1.088	.174
	55	3.498	1.189	1.077	.156
	60	3.176	1.182	1.064	.138
	65	2.837	1.104	1.051	.123
	70	2.487	1.073	1.036	.109
	75	2.085	1.042	1.021	.083
	80	1.647	1.010	1.005	.070
	85	1.240	.980	.990	.058
	90	.833	.951	.975	.045
	95	.495	.919	.950	.030
	100	.018	0	0	0

L. E. radius: 0.473 percent c

T. E. radius: 0.020 percent c

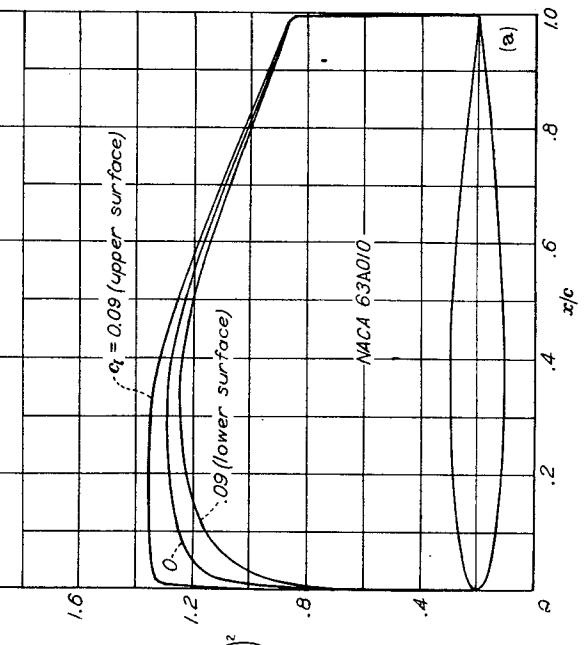
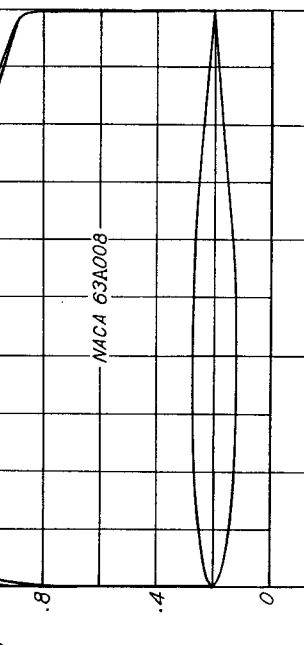
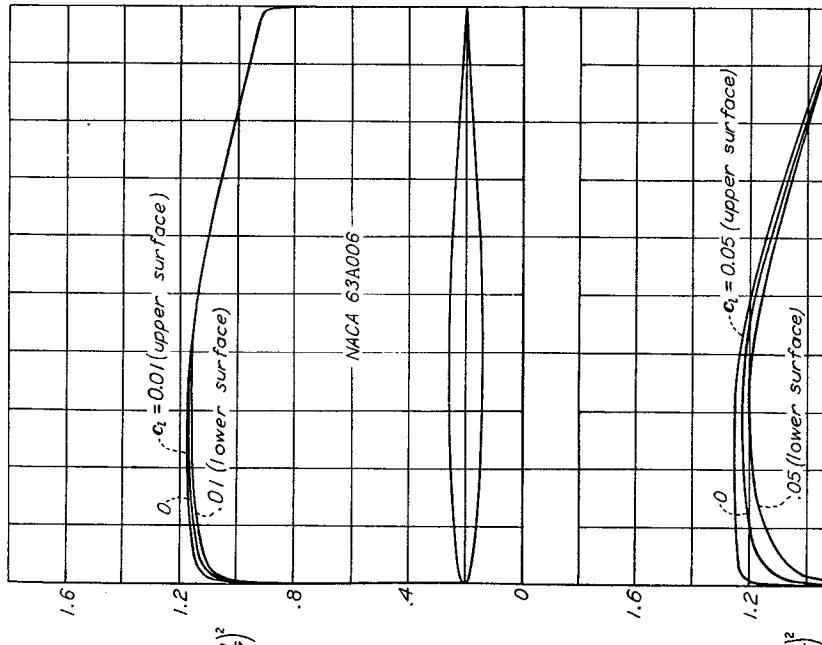
NACA 63A010 BASIC THICKNESS FORM

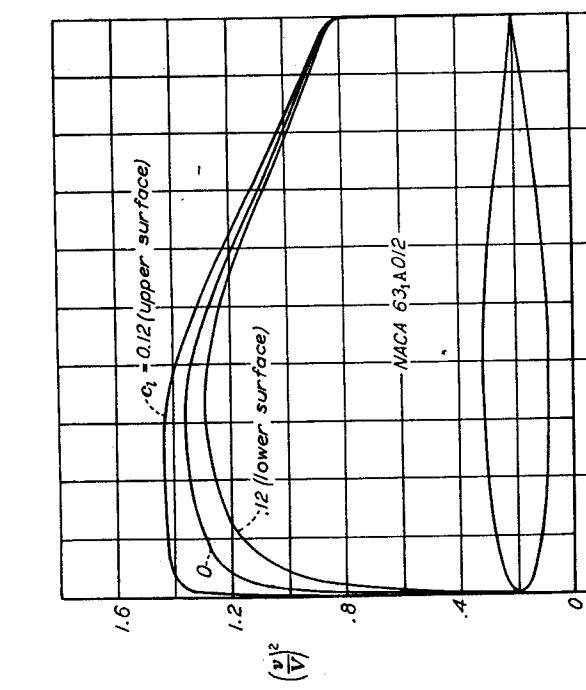
	$x$ (percent c)	$y$ (percent c)	$(v/V)^2$	$v/V$	$\Delta v_a/V$
	0	0	0	0	0
	.5	.816	.774	.880	2.805
	.75	.983	.985	.982	1.833
	1.25	1.240	1.035	1.121	1.504
	2.5	1.737	1.140	1.068	1.317
	5.0	2.412	1.200	1.095	.967
	7.5	2.917	1.225	1.107	.844
	10	3.324	1.245	1.116	.650
	15	3.950	1.268	1.126	.483
	20	4.460	1.282	1.132	.383
	25	4.714	1.290	1.136	.324
	30	4.913	1.294	1.138	.297
	35	4.995	1.291	1.136	.298
	40	4.988	1.279	1.131	.195
	45	4.927	1.258	1.122	.151
	50	4.613	1.230	1.109	.115
	55	4.311	1.196	1.094	.107
	60	3.943	1.162	1.078	.122
	65	3.517	1.125	1.061	.108
	70	3.044	1.086	1.042	.094
	75	2.545	1.048	1.024	.081
	80	2.040	1.009	1.004	.078
	85	1.535	.972	.986	.057
	90	1.030	.938	.969	.044
	95	.525	.900	.949	.030
	100	.021	0	0	0

L. E. radius: 0.742 percent c

T. E. radius: 0.023 percent c

FIGURE 2.—NACA 6A-series basic thickness forms.



NACA 63<sub>1</sub>A012 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(v/V)^2$		$v/V$	$\Delta v_a/V$
		0	$(v/V)^2$		
0	0	0.973	0.686	0	2.361
.5	1.173	1.024	.992	1.701	1.515
1.25	1.492	1.985	.992	1.288	1.365
2.5	2.078	1.337	1.136	1.066	.679
5.0	2.895	1.229	1.229	1.109	.539
10	3.504	1.295	1.295	1.125	.492
15	3.994	1.291	1.291	1.136	.394
20	4.747	1.324	1.151	1.151	.325
25	5.287	1.314	1.159	1.164	.291
30	5.864	1.335	1.166	1.166	.248
35	5.901	1.360	1.165	1.165	.219
40	5.945	1.337	1.165	1.165	.196
45	5.957	1.340	1.145	1.145	.174
50	5.792	1.312	1.058	1.058	.095
55	5.147	1.275	1.121	1.121	.669
60	5.148	1.234	1.150	1.150	.555
65	4.760	1.191	1.091	1.091	.120
70	4.186	1.145	1.070	1.070	.105
75	3.621	1.088	1.048	1.048	.092
80	3.026	1.031	1.025	1.025	.070
85	2.426	1.007	1.003	1.003	.066
90	1.826	1.064	.982	.982	.055
95	1.225	.925	.962	.962	.042
100	.625	0	.880	.880	.029

L. E. radius: 1.071 percent c  
T. E. radius: 0.028 percent c

NACA 63<sub>2</sub>A015 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(v/V)^2$		$v/V$	$\Delta v_a/V$
		0	$(v/V)^2$		
0	0	0	0	0	1.930
.5	1.203	1.550	1.550	.742	1.504
.75	1.448	1.825	1.825	1.168	1.176
1.25	1.844	1.882	1.882	1.058	.905
2.5	2.570	1.120	1.120	1.145	.174
5.0	3.618	1.275	1.275	1.129	.164
7.5	4.382	1.323	1.323	1.111	.136
10	4.997	1.361	1.361	1.091	.120
15	5.942	1.408	1.408	1.070	.105
20	6.619	1.437	1.437	1.050	.092
25	7.091	1.465	1.465	1.030	.070
30	7.384	1.464	1.464	1.016	.066
35	7.496	1.458	1.458	1.003	.055
40	7.435	1.435	1.435	1.000	.042
45	7.216	1.396	1.396	1.000	.042
50	6.868	1.349	1.349	1.011	.052
55	6.387	1.356	1.356	1.034	.134
60	5.820	1.237	1.237	1.118	.118
65	5.173	1.175	1.175	1.112	.104
70	4.468	1.115	1.115	1.084	.090
75	3.731	1.055	1.055	1.056	.077
80	2.991	1.000	1.000	1.000	.064
85	2.252	.975	.975	.975	.052
90	1.512	.990	.990	.949	.040
95	.772	.850	.850	.922	.028
100	.632	0	0	0	0

L. E. radius: 1.630 percent c  
T. E. radius: 0.037 percent c

NACA 63<sub>2</sub>A015 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(v/V)^2$		$v/V$	$\Delta v_a/V$
		0	$(v/V)^2$		
0	0	0	0	0	1.930
.5	.485	1.019	1.019	1.099	2.101
.75	.585	1.046	1.046	1.023	1.786
1.25	.739	1.076	1.076	1.037	1.422
2.5	1.016	1.106	1.106	1.052	.980
5.0	1.389	1.118	1.118	1.057	.694
7.5	1.684	1.126	1.126	1.061	.564
10	1.919	1.132	1.132	1.064	.482
15	2.283	1.141	1.141	1.088	.382
20	2.557	1.149	1.149	1.072	.321
25	2.737	1.154	1.154	1.074	.278
30	2.896	1.158	1.158	1.076	.246
35	2.977	1.162	1.162	1.078	.219
40	2.999	1.165	1.165	1.079	.197
45	2.945	1.156	1.156	1.075	.177
50	2.825	1.142	1.142	1.069	.159
55	2.653	1.125	1.125	1.061	.143
60	2.458	1.107	1.107	1.052	.126
65	2.188	1.087	1.087	1.043	.112
70	1.907	1.086	1.086	1.032	.099
75	1.692	1.043	1.043	1.021	.087
80	1.255	1.018	1.018	1.009	.074
85	.967	.992	.992	.986	.061
90	.659	.984	.984	.982	.047
95	.331	.965	.965	.967	.033
100	.013	0	0	0	0

L. E. radius: 0.246 percent c  
T. E. radius: 0.014 percent c

FIGURE 2.—Continued.

## NACA 64A008 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(v/V)^2$	$v/V$	$\Delta v_d/V$
0	0	0	0	3.546
.5	.646	.947	.973	.372
.75	.778	1.005	1.002	1.697
1.25	.983	1.008	1.003	1.392
2.5	1.353	1.122	1.059	.971
5.0	1.863	1.151	1.073	.692
7.5	2.245	1.165	1.079	.564
10	2.559	1.176	1.084	.481
15	3.047	1.191	1.091	.382
20	3.414	1.201	1.096	.323
25	3.681	1.209	1.100	.279
30	3.866	1.217	1.103	.247
35	3.972	1.221	1.105	.221
40	3.998	1.225	1.107	.198
45	3.921	1.211	1.100	.177
50	3.757	1.191	1.091	.158
55	3.524	1.167	1.080	.141
60	3.234	1.141	1.088	.125
65	2.897	1.113	1.055	.111
70	2.521	1.084	1.041	.098
75	2.117	1.053	1.026	.084
80	1.698	1.020	1.010	.072
85	1.278	.987	.983	.059
90	.858	.951	.975	.045
95	.438	.914	.966	.032
100	.018	0	0	0

L. E. radius: 0.459 percent c

T. E. radius: 0.020 percent c

## NACA 64A010 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(v/V)^2$	$v/V$	$\Delta v_d/V$
0	0	0	0	2.88
.5	.804	.868	.952	.845
.75	.969	1.052	1.021	1.303
1.25	1.225	1.130	1.083	1.085
2.5	1.688	1.178	1.085	.957
5.0	2.377	1.201	1.086	.688
7.5	2.805	1.201	1.086	.562
10	3.199	1.217	1.103	.480
15	3.813	1.238	1.113	.382
20	4.272	1.254	1.120	.324
25	4.666	1.266	1.125	.280
30	4.837	1.275	1.129	.248
35	4.968	1.282	1.132	.221
40	4.986	1.288	1.135	.199
45	4.894	1.268	1.126	.177
50	4.684	1.240	1.114	.158
55	4.388	1.208	1.099	.140
60	4.021	1.174	1.084	.124
65	3.507	1.139	1.087	.109
70	3.127	1.102	1.050	.096
75	2.623	1.063	1.031	.083
80	2.103	1.023	1.011	.070
85	1.582	.981	.990	.058
90	1.062	.938	.969	.044
95	.541	.838	.945	.031
100	.021	0	0	0

L. E. radius: 0.687 percent c

T. E. radius: 0.023 percent c

## NACA 64A012 BASIC THICKNESS FORM

 $c_t = 0.13$  (upper surface) $c_t = 0.13$  (lower surface)

(c)

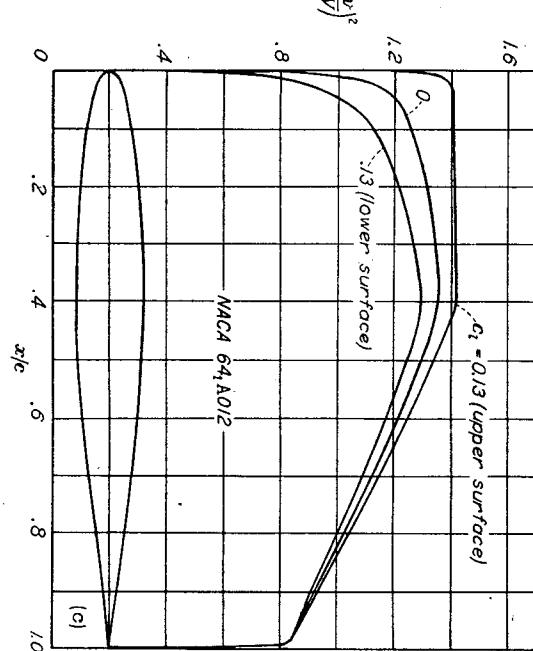
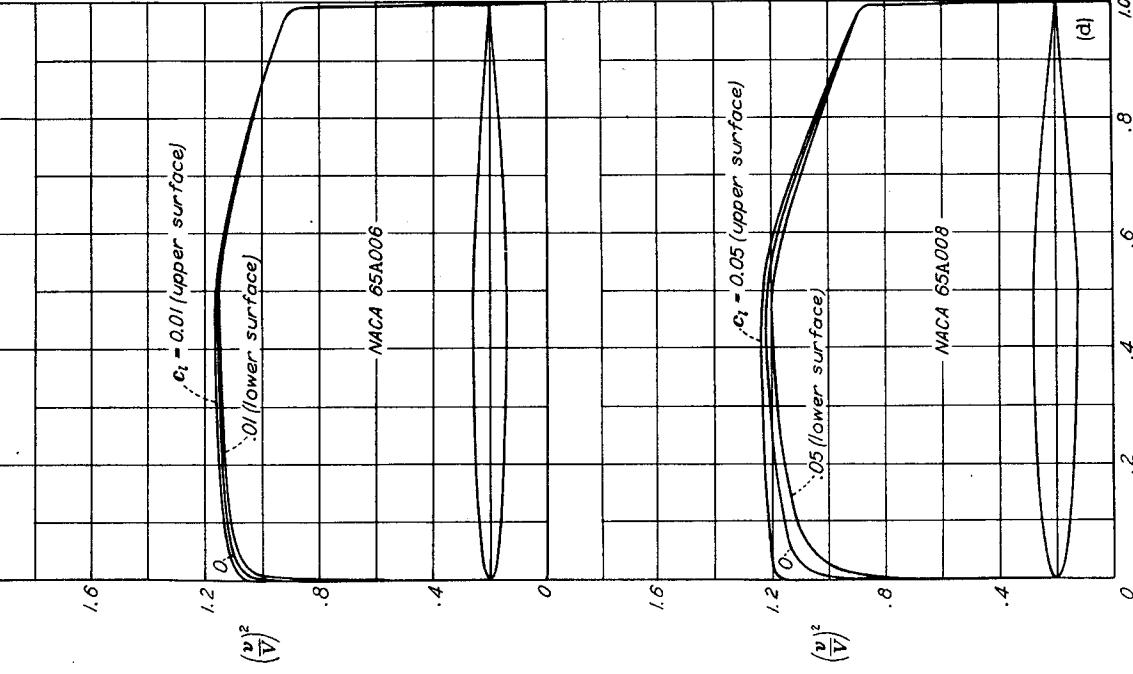
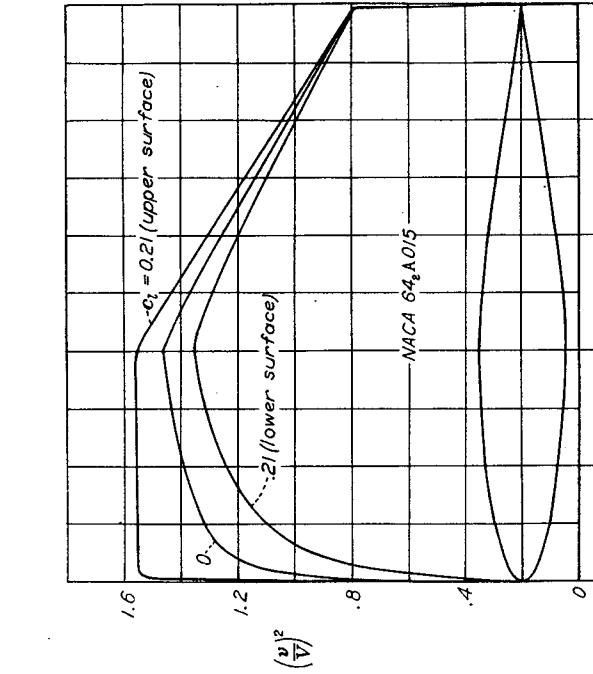


FIGURE 2.—Continued.



NACA 64A015 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(y/V)^2$	$v/V$	$\Delta v_a/V$
0	0	0	0	1.956
.5	.193	.678	.823	1.532
.75	.496	.789	.888	1.404
1.25	.815	.938	.967	1.189
2.5	2.508	1.110	.967	.912
5.0	3.477	1.226	1.077	.671
7.5	4.202	1.280	1.131	.552
10	4.798	1.314	1.146	.478
15	5.732	1.360	1.166	.384
20	6.423	1.390	1.179	.326
25	6.926	1.413	1.189	.283
30	7.270	1.430	1.196	.249
35	7.463	1.446	1.202	.222
40	7.487	1.458	1.207	.201
45	7.313	1.414	1.189	.177
50	6.978	1.364	1.168	.156
55	6.517	1.311	1.145	.137
60	5.956	1.255	1.120	.121
65	5.311	1.198	1.095	.106
70	4.690	1.139	1.067	.091
75	3.847	1.079	1.039	.078
80	3.084	1.020	1.010	.065
85	2.321	.961	.980	.053
90	1.558	.901	.949	.041
95	.785	.843	.918	.027
100	.032	0	0	0

 $L_c/E$  radius: 1.561 percent c $T_c/E$  radius: 0.037 percent c

NACA 65A006 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(y/V)^2$	$v/V$	$\Delta v_a/V$
0	0	0	0	4.879
.5	.464	1.034	1.017	2.145
.75	.563	1.033	1.021	1.763
1.25	.718	1.058	1.029	1.385
2.5	.981	1.087	1.039	.966
5.0	1.313	1.101	1.049	.688
7.5	1.591	1.112	1.055	.562
10	1.824	1.120	1.088	.490
15	2.194	1.131	1.083	.392
20	2.474	1.139	1.067	.323
25	2.687	1.145	1.070	.278
30	2.842	1.149	1.072	.246
35	2.945	1.153	1.074	.219
40	2.996	1.157	1.076	.198
45	2.992	1.159	1.077	.178
50	2.925	1.157	1.076	.161
55	2.793	1.141	1.088	.143
60	2.692	1.124	1.080	.127
65	2.364	1.106	1.052	.109
70	2.087	1.083	1.041	.099
75	1.775	1.059	1.029	.087
80	1.437	1.032	1.016	.076
85	1.083	1.003	1.001	.061
90	.727	.973	.988	.047
95	.370	.936	.967	.033
100	.013	0	0	0

 $L_c/E$  radius: 0.229 percent c $T_c/E$  radius: 0.014 percent c

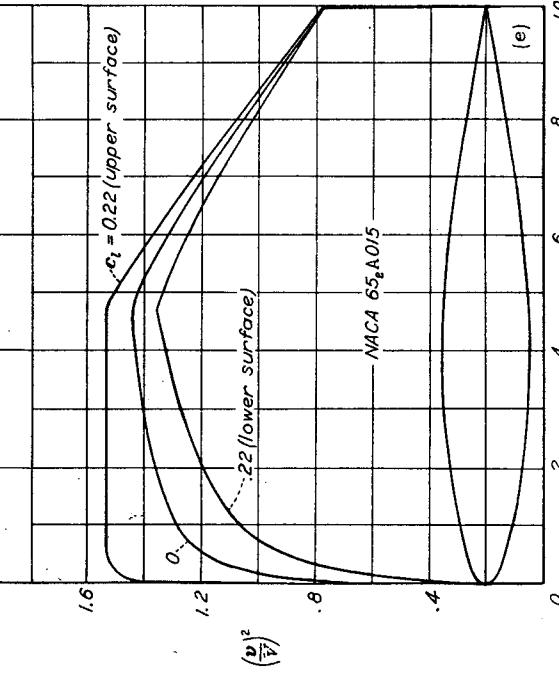
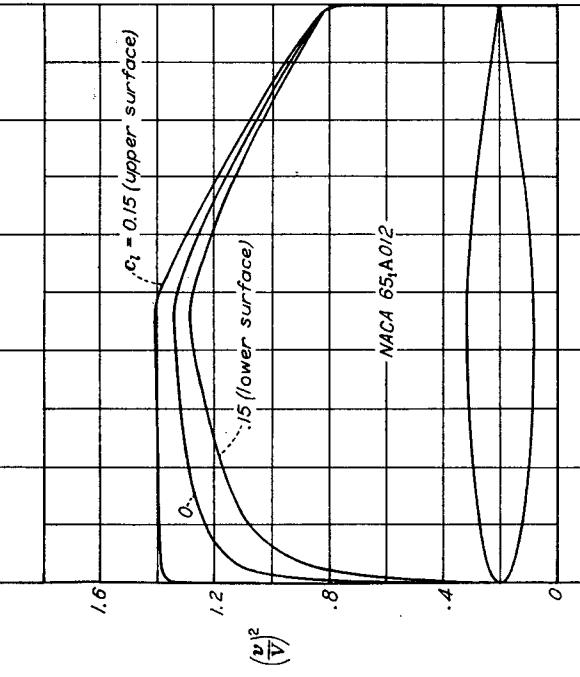
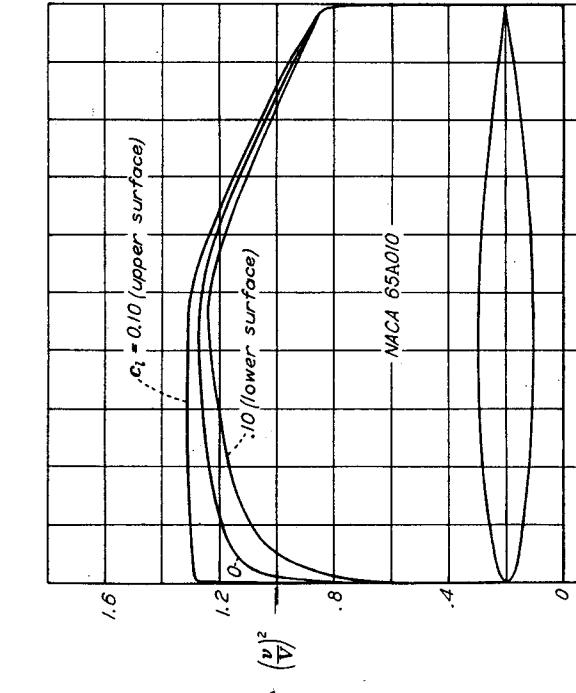
NACA 65A008 BASIC THICKNESS FORM

$x$ (percent c)	$y$ (percent c)	$(y/V)^2$	$v/V$	$\Delta v_a/V$
0	0	0	0	3.698
.5	.615	0.973	0.986	2.010
.75	.746	1.001	1.000	1.663
1.25	.951	1.038	1.010	1.323
2.5	1.303	1.088	1.045	.954
5.0	1.749	1.127	1.062	.685
7.5	2.120	1.145	1.070	.561
10	2.432	1.157	1.074	.479
15	2.926	1.175	1.084	.382
20	3.301	1.186	1.089	.322
25	3.585	1.195	1.093	.279
30	3.791	1.202	1.096	.247
35	3.928	1.207	1.098	.219
40	3.995	1.213	1.101	.198
45	3.988	1.217	1.103	.178
50	3.925	1.214	1.102	.161
55	3.714	1.191	1.091	.144
60	3.456	1.167	1.089	.128
65	3.135	1.139	1.112	.108
70	2.783	1.108	1.053	.086
75	2.348	1.076	1.037	.073
80	1.808	1.041	1.020	.050
85	1.430	1.002	1.001	.046
90	.960	.961	.980	.031
95	.489	.916	.957	0
100	.018	0	0	0

 $L_c/E$  radius: 0.408 percent c $T_c/E$  radius: 0.028 percent c

NACA 65A008 BASIC THICKNESS FORM

Figure 2.—Continued.



	$x$ (percent $c$ )	$y$ (percent $c$ )	$(v/V)^2$	$v/V$	$\Delta v_d/V$
NACA 6A010 BASIC THICKNESS FORM					
0	0	0	.807	.947	2.987
	.5	.765	.948	.974	1.878
	.75	.998	1.000	1.005	1.303
	1.25	1.183	1.049	1.044	.936
	2.5	1.623	1.148	1.071	.679
	5.0	2.182	1.176	1.084	.559
	7.5	2.680	1.194	1.093	.478
	10	3.040	1.218	1.104	.382
	15	3.658	1.241	1.111	.323
	20	4.127	1.247	1.117	.281
	25	4.483	1.257	1.121	.249
	30	4.742	1.265	1.125	.222
	35	4.912	1.272	1.128	.198
	40	4.986	1.277	1.130	.178
	45	4.983	1.271	1.127	.161
	50	4.983	1.241	1.114	.144
	55	4.632	1.208	1.099	.127
	60	4.304	1.172	1.082	.111
	65	3.869	1.133	1.064	.097
	70	3.452	1.091	1.045	.084
	75	2.912	1.047	1.023	.071
	80	2.352	.999	.999	.058
	85	1.771	.949	.974	.046
	90	1.188	.883	.945	.029
	95	.604	0	0	0
	100	.021	0	0	0

L. E. radius: 0.639 percent  $c$ T. E. radius: 0.023 percent  $c$ 

	$x$ (percent $c$ )	$y$ (percent $c$ )	$(v/V)^2$	$v/V$	$\Delta v_d/V$
NACA 65A012 BASIC THICKNESS FORM					
0	0	0	.824	.908	2.520
	.5	.943	.883	.940	1.543
	1.25	1.106	.969	.984	1.263
	2.5	1.414	1.081	1.040	.914
	5.0	1.942	1.166	1.080	.672
	7.5	2.614	1.204	1.097	.557
	10	3.176	1.228	1.108	.477
	15	4.382	1.263	1.124	.382
	20	4.936	1.285	1.134	.324
	25	5.383	1.301	1.141	.281
	30	5.683	1.313	1.146	.250
	35	5.907	1.324	1.151	.224
	40	5.995	1.322	1.154	.198
	45	5.977	1.328	1.157	.178
	50	5.828	1.329	1.158	.161
	55	5.584	1.292	1.137	.143
	60	5.143	1.251	1.118	.126
	65	4.684	1.204	1.097	.111
	70	4.091	1.156	1.075	.096
	75	3.467	1.104	1.051	.082
	80	2.798	1.051	1.025	.069
	85	2.106	.994	.997	.057
	90	1.413	.936	.967	.043
	95	.719	.871	.933	.027
	100	.025	0	0	0

L. E. radius: 0.639 percent  $c$ T. E. radius: 0.023 percent  $c$ 

	$x$ (percent $c$ )	$y$ (percent $c$ )	$(v/V)^2$	$v/V$	$\Delta v_d/V$
NACA 65A015 BASIC THICKNESS FORM					
0	0	0	0	0	0
	.5	1.131	.714	.945	2.048
	.75	1.371	.781	.984	1.386
	1.25	1.750	.891	.994	1.417
	2.5	2.412	1.059	1.029	1.195
	5.0	3.285	1.187	1.089	.880
	7.5	3.982	1.245	1.115	.660
	10	4.563	1.280	1.131	.476
	15	5.488	1.328	1.152	.382
	20	6.198	1.350	1.166	.326
	25	6.734	1.383	1.176	.292
	30	7.122	1.401	1.184	.252
	35	7.376	1.416	1.190	.227
	40	7.496	1.427	1.195	.194
	45	7.467	1.437	1.199	.161
	50	7.269	1.419	1.191	.142
	55	6.903	1.388	1.170	.124
	60	6.393	1.311	1.145	.109
	65	5.772	1.249	1.118	.094
	70	5.083	1.186	1.089	.076
	75	4.282	1.123	1.056	.060
	80	3.451	1.056	1.028	.047
	85	2.508	.986	.983	.035
	90	1.743	.913	.956	.041
	95	.887	.841	.917	.026
	100	.032	0	0	0

L. E. radius: 1.446 percent  $c$ T. E. radius: 0.038 percent  $c$

**Mean line.**—In order that the addition of camber not change the pressure gradients over the basic thickness form, a mean line should be used which causes uniform load to be carried from the leading edge to a point at least as far back as the position of minimum pressure on the basic thickness form. The usual practice is to camber NACA 6-series airfoil sections with the  $a=1.0$  type of mean line because this mean line appears to be best for high maximum lift coefficients and, contrary to theoretical predictions, does not cause excessive quarter-chord pitching-moment coefficients.

The  $a=1.0$  type mean line was not considered desirable, however, for the NACA 6A-series basic thickness forms because the surfaces of the cambered airfoil sections would be curved near the trailing edge. The type of mean line best suited for maintaining straight sides on these newer sections would be one that is straight from 80 percent chord to the trailing edge. Such a camber line could be obtained by modifying an  $a=0.7$  mean line. Consideration of the effect of mean-line loading upon the maximum lift coefficient indicated, however, that a mean line having a uniform load distribution as far back along the chord as possible was desirable. It was found that the  $a=0.8$  type mean line could be made straight from approximately 85 percent chord to the trailing edge without causing a sharp break in the mean line and with very little curvature between the 80-percent- and 85-percent-chord stations. The aerodynamic advantages of using this mean line in preference to one having uniform load to 70 percent chord were considered to be more important than the slight curvature existing in the modified  $a=0.8$  mean line. For this reason, all cambered NACA 6A-series airfoil sections have employed the  $a=0.8$  (modified) mean line.

The ordinates and load-distribution data corresponding to a design lift coefficient of 1.0 are presented in figure 3 for the  $a=0.8$  (modified) mean line. The ordinates of a mean line having any arbitrary design lift coefficient may be obtained simply by multiplying the ordinates presented by the desired design lift coefficient.

**Cambered airfoils.**—The method used for cambering the basic thickness distributions of figure 2 with the mean line of figure 17 is described and discussed in references 1 and 2. It consists essentially in laying out the ordinates of the basic thickness forms normal to the mean line at corresponding stations. A discussion of the method employed for combining the theoretical pressure-distribution data, presented in figures 2 and 3 for the mean-line and basic-thickness distributions, to give the approximate theoretical pressure distribution about a cambered or symmetrical airfoil section at any lift coefficient is given in reference 1.

#### APPARATUS AND TESTS

**Wind tunnel.**—All the tests described herein were conducted in the Langley two-dimensional low-turbulence pressure tunnel. The test section of this tunnel measures 3 feet by 7.5 feet. The models completely spanned the 3-foot dimension with the gaps between the model and tunnel

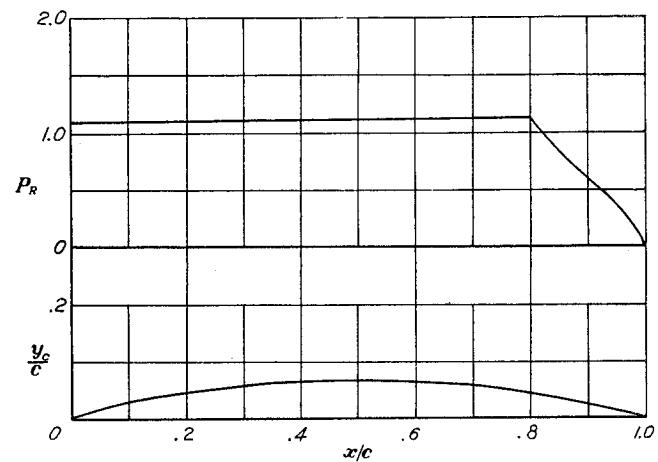


FIGURE 3.—Data for NACA mean line  $a=0.8$  (modified).

walls sealed to prevent air leakage. Lift measurements were made by taking the difference between the pressure reaction upon the floor and ceiling of the tunnel, drag results were obtained by the wake-survey method, and pitching moments were determined with a torque balance. A more complete description of the tunnel and the method of obtaining and reducing the data are contained in reference 1.

**Models.**—The seven airfoil sections for which the experimental aerodynamic characteristics were obtained are:

- NACA 63A010
- NACA 63A210
- NACA 64A010
- NACA 64A210, NACA 64<sub>1</sub>A212, NACA 64<sub>2</sub>A215
- NACA 64A410

The models representing the airfoil sections were of 24-inch chord and were constructed of laminated mahogany. The models were painted with lacquer and then sanded with No. 400 carborundum paper until aerodynamically smooth surfaces were obtained. The ordinates of the models tested are presented in tables I to VII.

## THEORETICAL AND EXPERIMENTAL DATA FOR A NUMBER OF NACA 6A-SERIES AIRFOIL SECTIONS

TABLE I.—ORDINATES OF NACA 63A010 AIRFOIL SECTION

[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.5	.816	.5	-.816
.75	.983	.75	-.983
1.25	1.250	1.25	-.1.250
2.5	1.737	2.5	-.1.737
5.0	2.412	5.0	-.2.412
7.5	2.917	7.5	-.2.917
10	3.324	10	-.3.324
15	3.950	15	-.3.950
20	4.400	20	-.4.400
25	4.714	25	-.4.714
30	4.913	30	-.4.913
35	4.995	35	-.4.995
40	4.968	40	-.4.968
45	4.837	45	-.4.837
50	4.613	50	-.4.613
55	4.311	55	-.4.311
60	3.943	60	-.3.943
65	3.517	65	-.3.517
70	3.044	70	-.3.044
75	2.545	75	-.2.545
80	2.040	80	-.2.040
85	1.535	85	-.1.535
90	1.030	90	-.1.030
95	.525	95	-.525
100	.021	100	-.021

L. E. radius: 0.742  
T. E. radius: 0.023

TABLE III.—ORDINATES OF NACA 64A010 AIRFOIL SECTION

[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.5	.804	.5	-.804
.75	.969	.75	-.969
1.25	1.225	1.25	-.1.225
2.5	1.688	2.5	-.1.688
5.0	2.327	5.0	-.2.327
7.5	2.805	7.5	-.2.805
10	3.199	10	-.3.199
15	3.813	15	-.3.813
20	4.272	20	-.4.272
25	4.606	25	-.4.606
30	4.837	30	-.4.837
35	4.968	35	-.4.968
40	4.995	40	-.4.995
45	4.894	45	-.4.894
50	4.684	50	-.4.684
55	4.388	55	-.4.388
60	4.021	60	-.4.021
65	3.597	65	-.3.597
70	3.127	70	-.3.127
75	2.623	75	-.2.623
80	2.103	80	-.2.103
85	1.582	85	-.1.582
90	1.062	90	-.1.062
95	.541	95	-.541
100	.021	100	-.021

L. E. radius: 0.687  
T. E. radius: 0.023

TABLE II.—ORDINATES OF NACA 63A210 AIRFOIL SECTION

[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.423	.868	.577	-.756
.684	1.058	.826	-.900
1.151	1.367	1.349	-.1.125
2.384	1.944	2.616	-.1.522
4.869	2.769	5.131	-.2.047
7.364	3.400	7.636	-.2.428
9.863	3.917	10.137	-.2.725
14.869	4.729	15.131	-.3.167
19.882	5.328	20.118	-.3.468
24.898	5.764	25.102	-.3.662
29.916	6.060	30.084	-.3.761
34.935	6.219	35.065	-.3.771
39.955	6.247	40.045	-.3.689
44.975	6.151	45.025	-.3.523
49.994	5.943	50.006	-.3.283
55.012	5.637	54.988	-.2.985
60.028	5.245	59.972	-.2.641
65.041	4.772	64.959	-.2.262
70.052	4.227	69.948	-.1.861
75.061	3.624	74.939	-.1.464
80.074	2.974	79.926	-.1.104
85.072	2.254	84.928	-.812
90.050	1.519	89.950	-.539
95.026	.789	94.974	-.279
100.000	.021	100.000	-.021

L. E. radius: 0.742  
T. E. radius: 0.023  
Slope of radius through L. E.: 0.095

TABLE IV.—ORDINATES OF NACA 64A210 AIRFOIL SECTION

[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.424	.856	.576	-.744
.665	1.044	.835	-.886
1.153	1.342	1.347	-.1.100
2.387	1.895	2.613	-.1.473
4.874	2.685	5.126	-.1.963
7.369	3.288	7.631	-.2.316
9.868	3.792	10.132	-.2.600
14.874	4.592	15.126	-.3.030
19.885	5.200	20.115	-.3.340
24.900	5.656	25.100	-.3.554
29.917	5.984	30.083	-.3.688
34.935	6.192	35.065	-.3.744
39.955	6.274	40.045	-.3.716
44.975	6.208	45.025	-.3.580
49.994	6.014	50.006	-.3.354
55.012	6.714	54.988	-.3.062
60.028	5.323	59.972	-.2.719
65.042	4.852	64.958	-.2.342
70.054	4.310	69.946	-.1.944
75.063	3.702	74.937	-.1.542
80.076	3.037	79.924	-.1.167
85.074	2.301	84.926	-.1.559
90.052	1.551	89.948	-.1.571
95.027	.785	94.974	-.2.295
100.000	.021	100.000	-.021

L. E. radius: 0.687  
T. E. radius: 0.023  
Slope of radius through L. E.: 0.095

TABLE V.—ORDINATES OF NACA 64A410 AIRFOIL SECTION

[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.350	.902	.650	-.678
.582	1.112	.918	-.796
1.059	1.451	1.441	-.969
2.276	2.095	2.724	-.151
4.749	3.034	5.251	-.592
7.230	3.865	7.770	-.191
9.737	4.380	10.263	-.196
14.748	5.366	15.252	-.244
19.770	6.126	20.230	-.246
24.800	6.705	25.200	-.249
29.834	7.131	30.166	-.253
34.871	7.414	35.129	-.258
39.910	7.552	40.090	-.246
44.950	7.522	45.050	-.266
49.989	7.344	50.011	-.224
55.025	7.040	54.975	-.136
60.057	6.624	59.943	-.118
65.085	6.106	64.915	-.086
70.108	5.490	69.892	-.760
75.126	4.780	74.874	-.460
80.151	3.967	79.849	-.229
85.148	3.018	84.852	-.132
90.104	2.038	89.896	-.076
95.053	1.028	94.947	-.048
100.000	.021	100.000	-.021

L. E. radius: 0.687  
T. E. radius: 0.023  
Slope of radius through L. E.: 0.190

TABLE VI.—ORDINATES OF NACA 641A212 AIRFOIL SECTION

[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.409	1.013	.591	-.901
.648	1.233	.852	-.1075
1.135	1.580	1.365	-.1338
2.365	2.225	2.635	-.1803
4.849	3.145	5.151	-.2423
7.343	3.846	7.657	-.2874
9.842	4.432	10.158	-.3240
14.849	5.358	15.151	-.3796
19.862	6.080	20.138	-.4200
24.880	6.584	25.120	-.4482
29.900	6.956	30.100	-.4660
34.922	7.189	35.078	-.4741
39.946	7.272	40.054	-.4714
44.970	7.177	45.030	-.4549
49.993	6.935	50.007	-.4275
55.015	6.570	54.985	-.3918
60.034	6.103	59.966	-.3499
65.050	5.544	64.950	-.3034
70.064	4.903	69.936	-.2537
75.075	4.197	74.925	-.2037
80.090	3.433	79.910	-.1563
85.088	2.601	84.912	-.159
90.062	1.751	89.938	-.771
95.032	.888	94.968	-.398
100.000	.025	100.000	-.025

L. E. radius: 0.904  
T. E. radius: 0.028  
Slope of radius through L. E.: 0.095

TABLE VII.—ORDINATES OF NACA 64A215 AIRFOIL SECTION

[Stations and ordinates given in percent of airfoil chord]

Upper surface		Lower surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.388	1.243	.612	-.131
.624	1.509	.876	-.351
1.107	1.930	1.393	-.688
2.333	2.713	2.667	-.291
4.811	3.833	5.189	-.311
7.304	4.683	7.696	-.711
9.802	5.391	10.198	-.199
14.811	6.510	15.189	-.498
19.827	7.351	20.173	-.491
24.849	7.975	25.151	-.873
29.876	8.417	30.125	-.612
34.903	8.686	35.097	-.628
39.933	8.766	40.067	-.208
44.963	8.627	45.037	-.599
49.992	8.308	50.008	-.648
55.018	7.843	54.982	-.191
60.042	7.258	59.958	-.654
65.063	6.566	64.937	-.086
70.079	5.782	69.921	-.346
75.093	4.926	74.907	-.2766
80.111	4.017	79.889	-.2147
85.109	3.039	84.891	-.1597
90.076	2.046	89.924	-.1066
95.039	1.039	94.961	-.549
100.000	.032	100.000	-.032

L. E. radius: 1.561  
T. E. radius: 0.037  
Slope of radius through L. E.: 0.095

**Tests.**—The tests of each smooth airfoil section consisted in measurements of the lift, drag, and quarter-chord pitching-moment coefficients at Reynolds numbers of  $3 \times 10^6$ ,  $6 \times 10^6$ , and  $9 \times 10^6$ . In addition, the lift and drag characteristics of each section were determined at a Reynolds number of  $6 \times 10^6$  with standard roughness applied to the leading edge of the model. The standard roughness employed on these 24-inch-chord models consisted of 0.011-inch-diameter carbide grains spread over a surface length of 8 percent of the chord back from the leading edge on the upper and lower surfaces. The grains were thinly spread to cover from 5 to 10 percent of this area. In an effort to obtain some idea of the effectiveness of the airfoil sections when equipped with trailing-edge high-lift devices, each section was fitted with a simulated split flap deflected 60°. Lift measurements with the split flap were made at a Reynolds number of  $6 \times 10^6$  with the airfoil leading edge both smooth and rough.

## RESULTS

The results obtained from tests of the seven airfoil sections are presented in figures 4 to 10 in the form of standard aerodynamic coefficients representing the lift, drag, and quarter-chord pitching-moment characteristics of the airfoil sections.

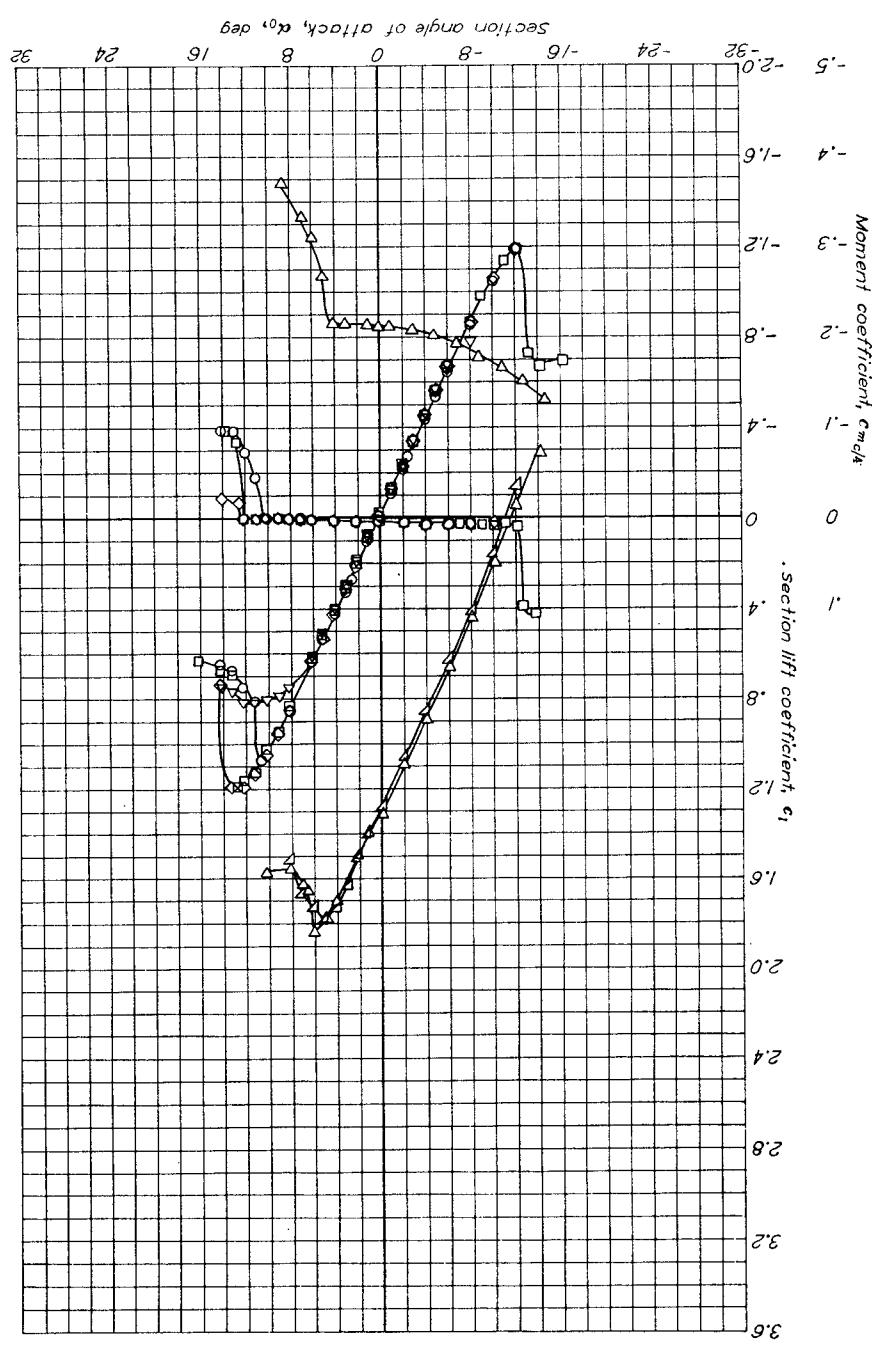
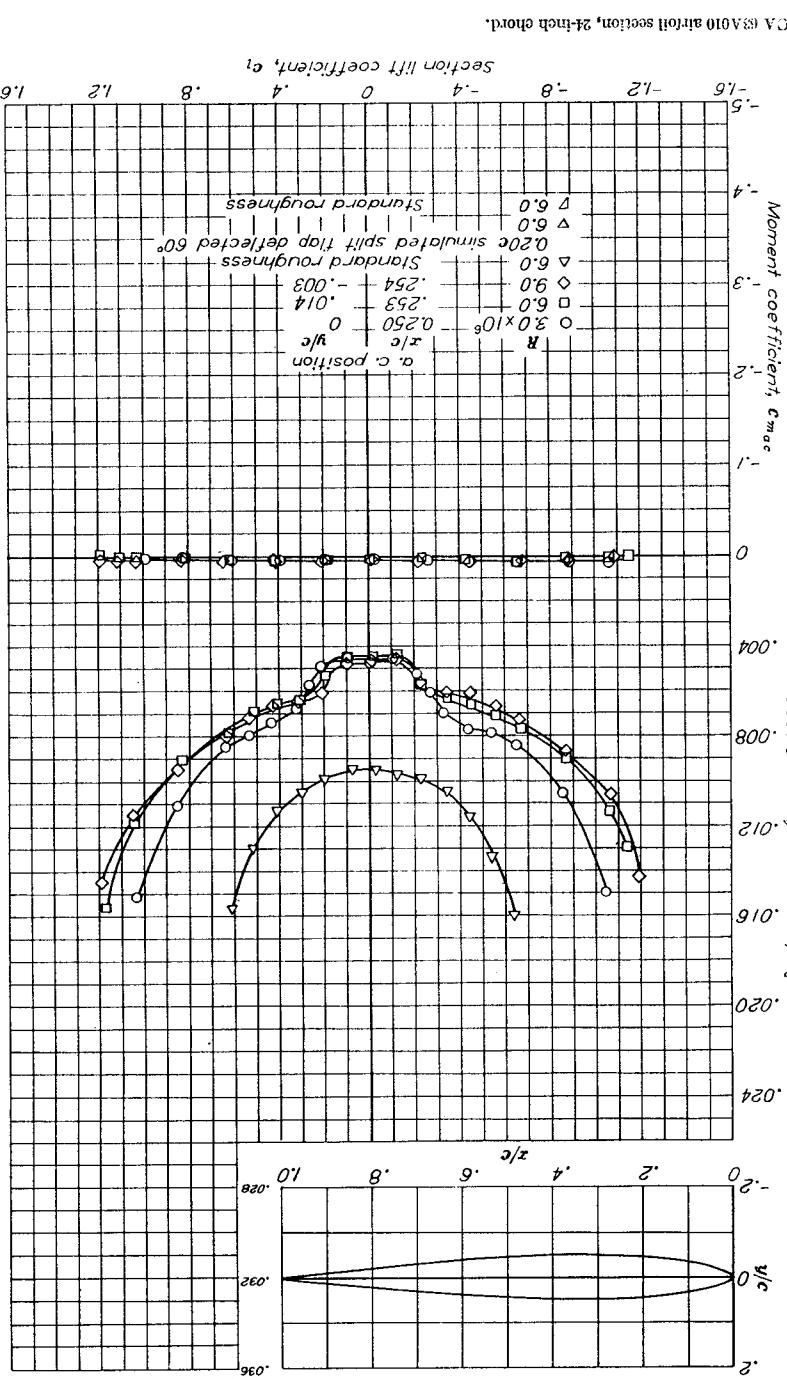
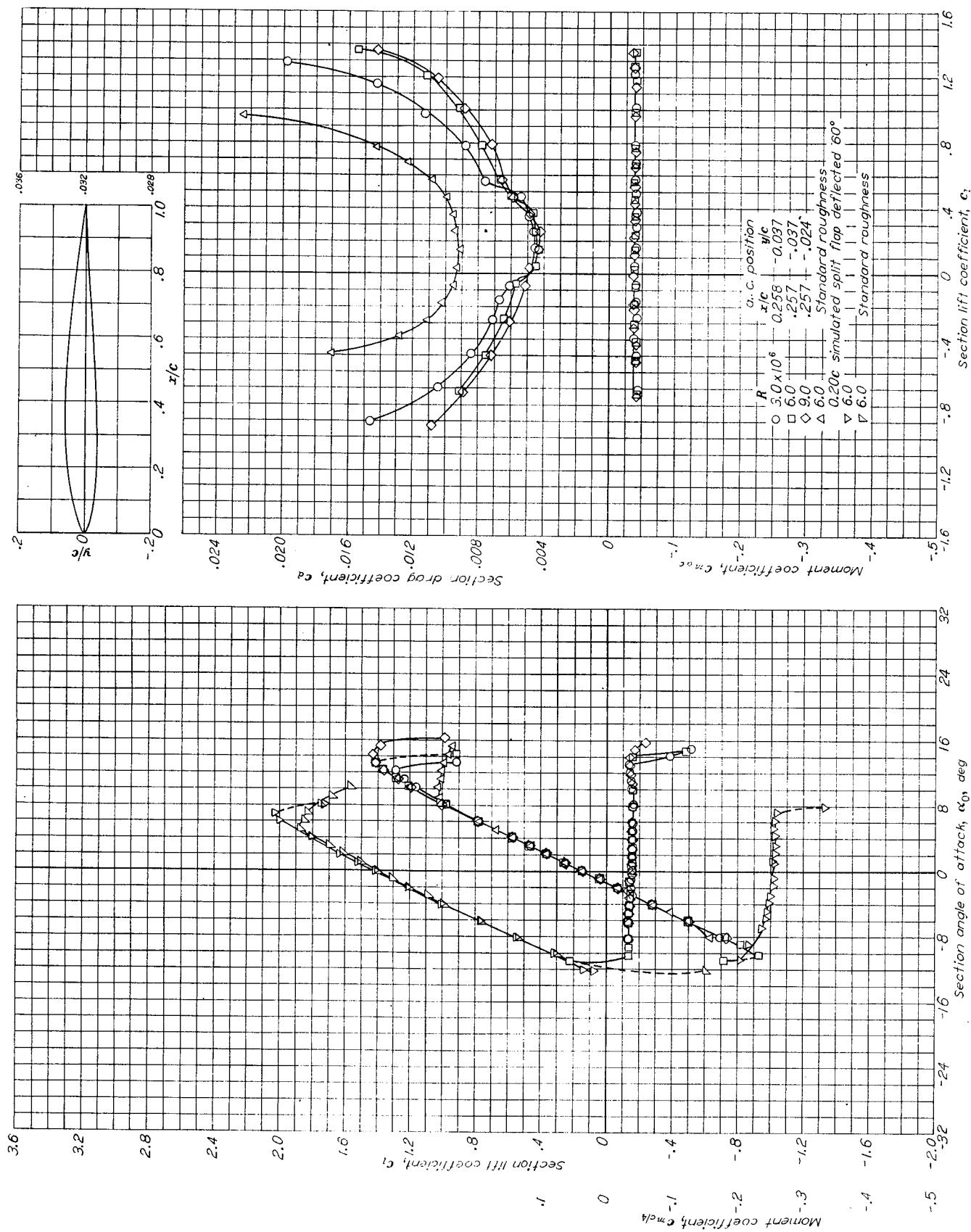


FIGURE 4.—Aerodynamic characteristics of the NACA 6A010 airfoil section, 24-inch chord.



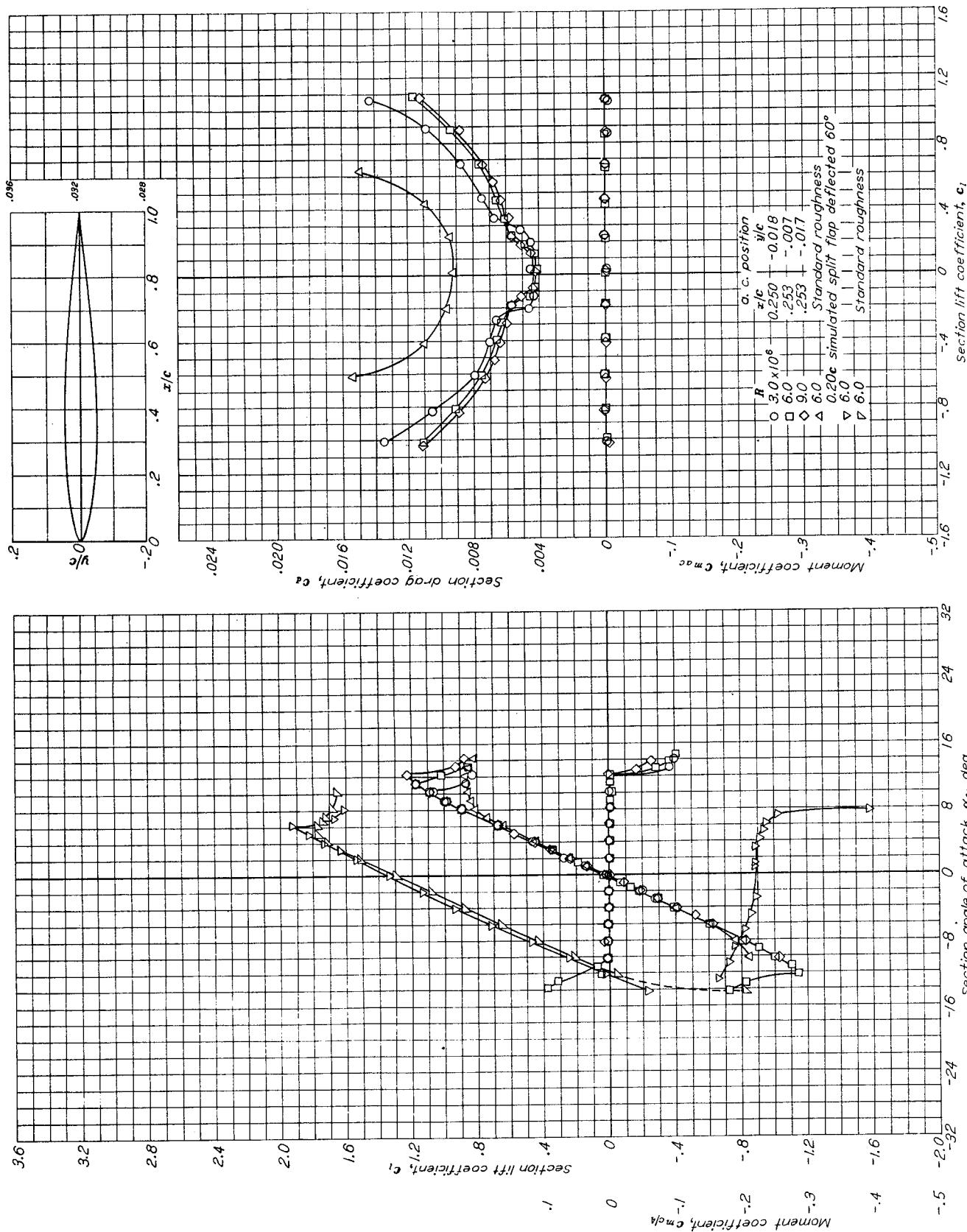


FIGURE 6.—Aerodynamic characteristics of the NACA 64A010 airfoil section, 24-inch chord.

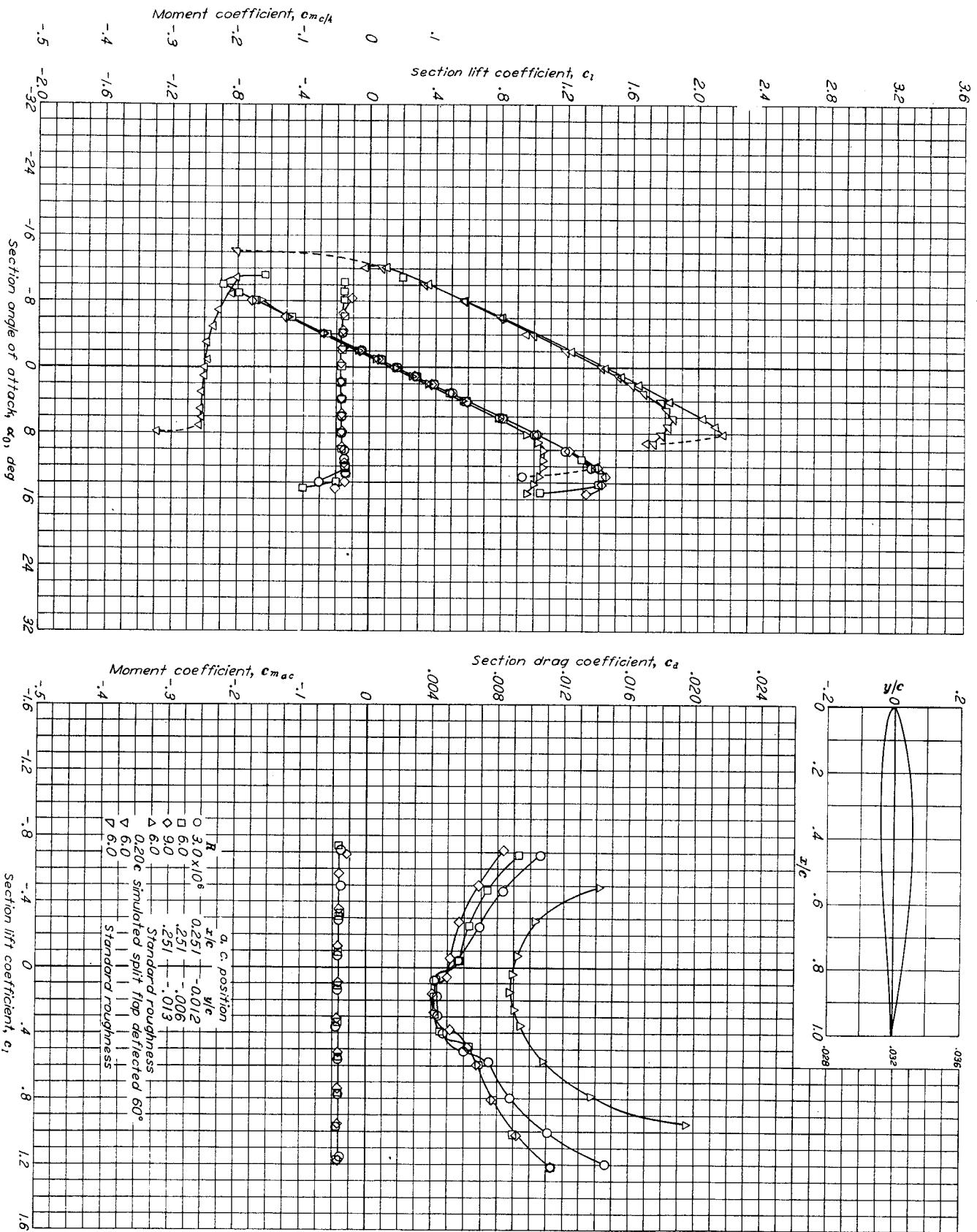


FIGURE 7.—Aerodynamic characteristics of the NACA 6A210 airfoil section, 24-inch chord.

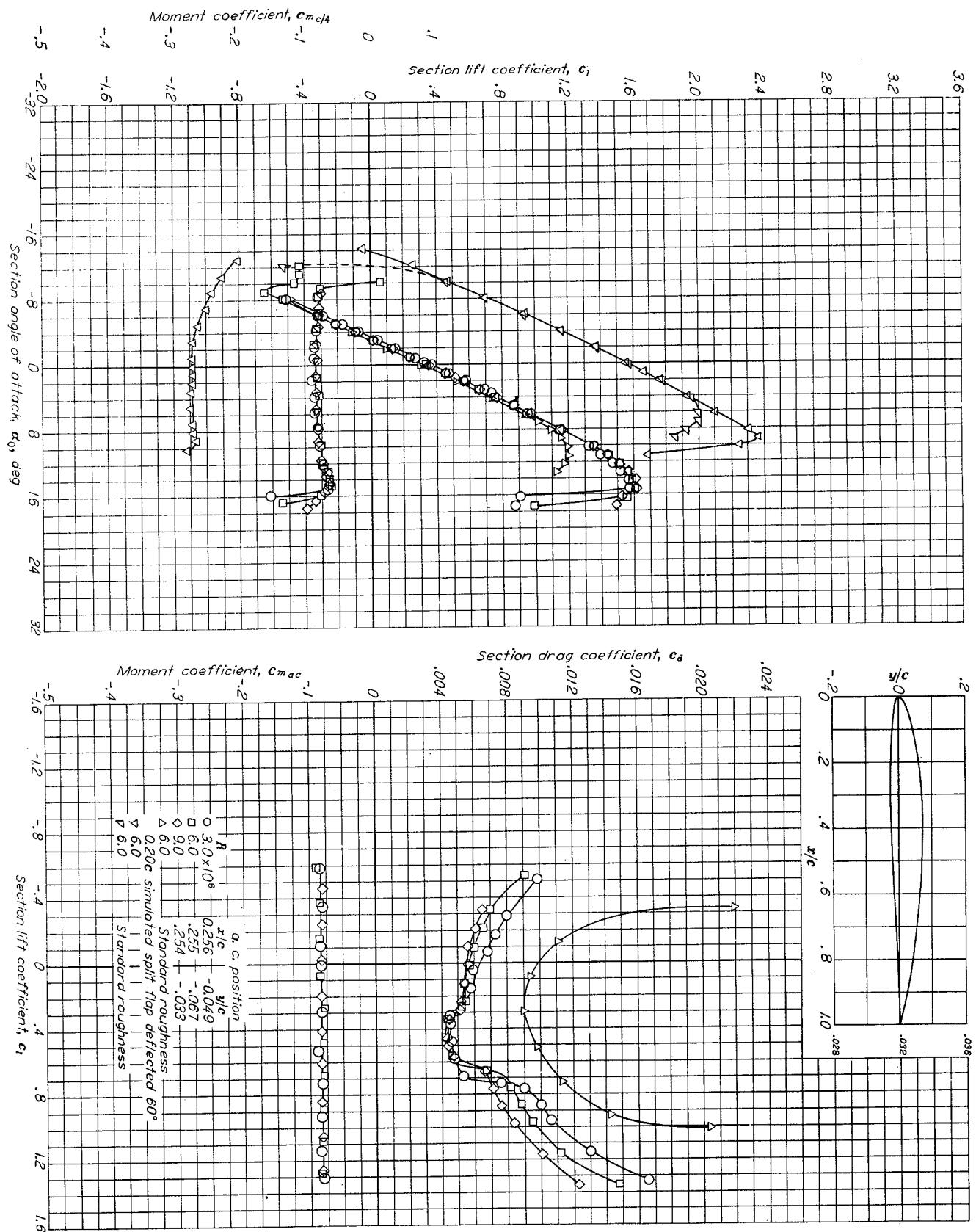


FIGURE 8.—Aerodynamic characteristics of the NACA 6A410 airfoil section, 24-inch chord.

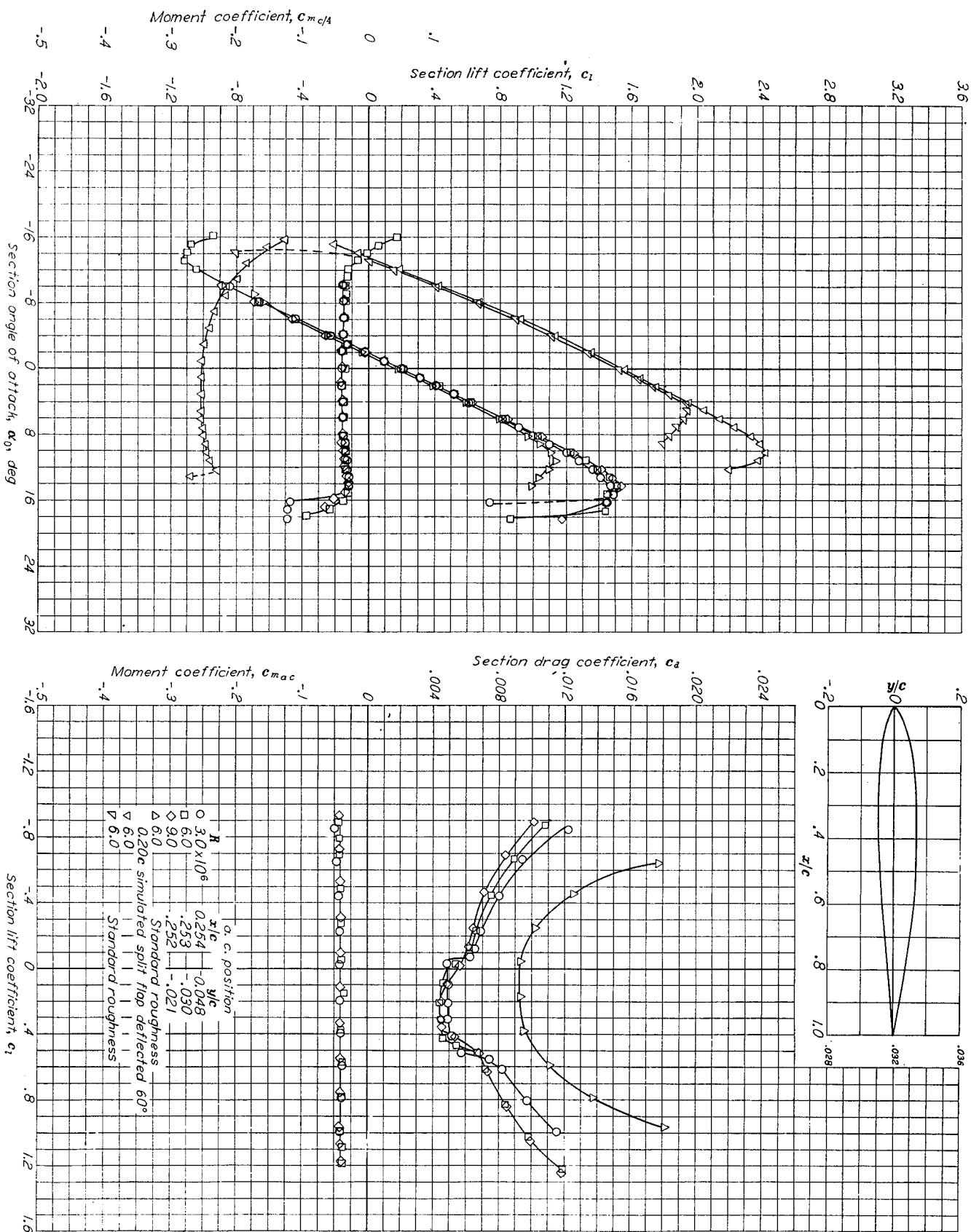


FIGURE 9.—Aerodynamic characteristics of the NACA 64A212 airfoil section, 24-inch chord.

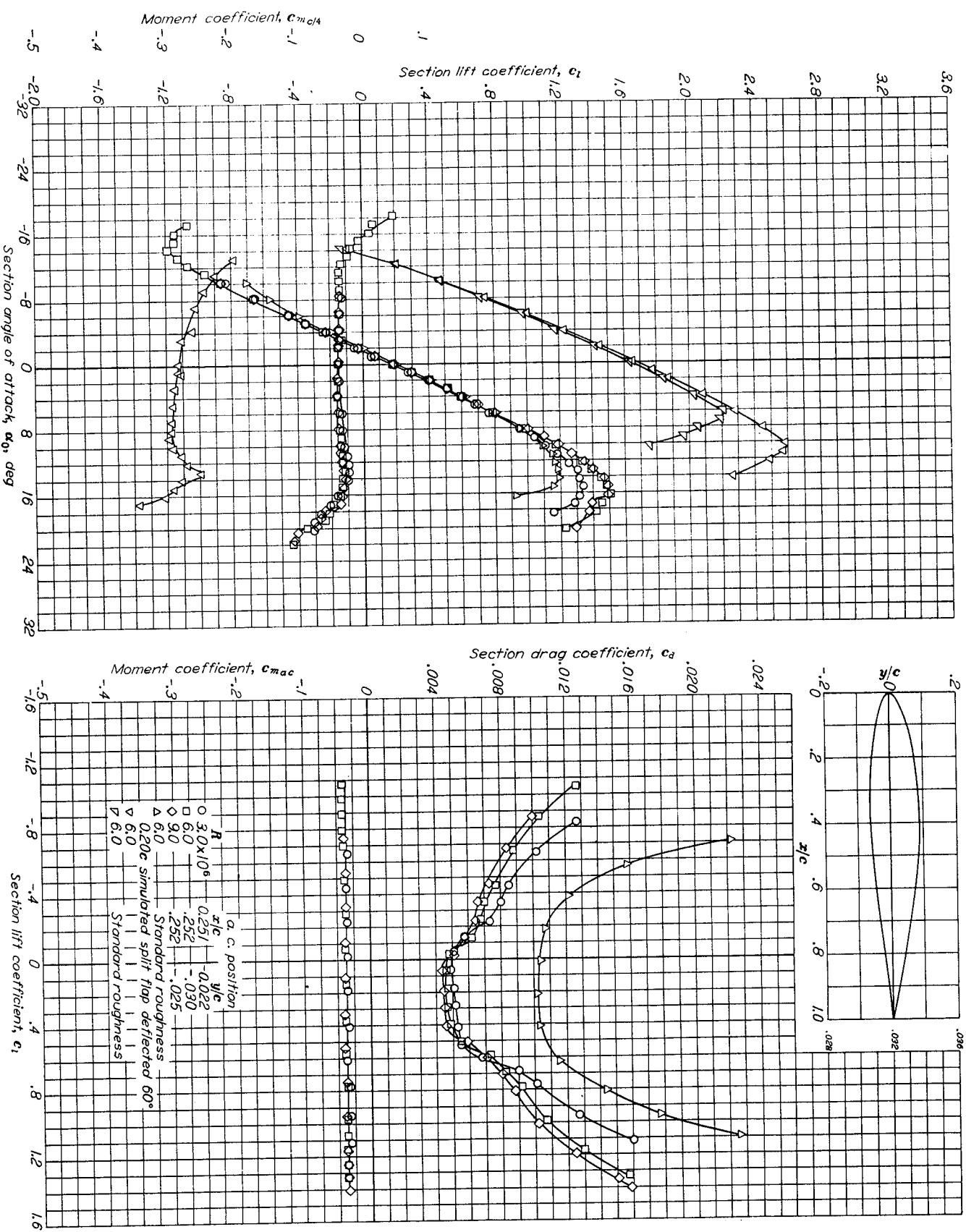


FIGURE 10.—Aerodynamic characteristics of the NACA 64A215 airfoil section, 24-inch chord.

The calculated position of the aerodynamic center and the variation of the pitching-moment coefficient with lift coefficient about this point are also included in these data. The influence of the tunnel boundaries has been removed from all the aerodynamic data by means of the following equations (developed in reference 1):

$$c_d = 0.990 c_d'$$

$$c_l = 0.973 c_l'$$

$$c_{m_{c/4}} = 0.951 c_{m_{c/4}'}$$

$$\alpha_0 = 1.015 \alpha_0'$$

where the primed quantities denote the measured coefficients.

### DISCUSSION

Although the amount of systematic aerodynamic data presented for NACA 6A-series airfoil sections is not large, it is enough to indicate the relative merits of the NACA 6A-series airfoil sections as compared with the NACA 6-series sections. The variation of the important aerodynamic characteristics of the five NACA 64A-series airfoils with the pertinent geometrical parameters of the airfoils is shown in figures 11 to 17, together with comparable data for NACA 64-series airfoils. The curves shown in figures 11 to 17 are for the NACA 64-series airfoil sections and are taken from the faired data of reference 1. The experimental points which appear on these figures represent the results obtained for the NACA 64A-series airfoil sections in the present investigation. Since only two NACA 63A-series sections were tested, comparative results are not presented for them. The effect of removing the cusp from the NACA 63-series

sections is about the same as that of removing the cusp from the NACA 64-series sections.

The comparative data showing the effects upon the aerodynamic characteristics of removing the trailing-edge cusp from NACA 6-series airfoil sections should be used with caution if the cusp removal is affected in some manner other than that indicated earlier in this paper. For example, if the cusp should be removed from a cambered airfoil by means of a straight-line fairing of the airfoil surfaces, the amount of camber would be decreased near the trailing edge. Naturally the effect upon the aerodynamic characteristics of removing the cusp in such a manner would not be the same as indicated by the comparative results presented for NACA 6-series and 6A-series airfoils.

**Drag.**—The variation of section minimum drag coefficient with airfoil thickness ratio at a Reynolds number of  $6 \times 10^6$  is shown in figure 11 for NACA 64-series and NACA 64A-series airfoil sections of various cambers, both smooth and with standard leading-edge roughness. As with the NACA 64-series sections (reference 1), the minimum drag coefficients of the NACA 64A-series sections show no consistent variation with camber. Comparison of the data of figure 11 indicates that removing the cusp from the trailing edge has no appreciable effect upon the minimum drag coefficients of the airfoils, either smooth or with standard leading-edge roughness.

Increasing the Reynolds number from  $3 \times 10^6$  to  $9 \times 10^6$  has about the same effect upon the minimum drag coefficient of NACA 64A-series airfoils (figs. 4 to 10) as that indicated in reference 1 for the NACA 64-series airfoils.

Some differences exist in the drag coefficients of NACA 64- and 64A-series airfoils outside the low-drag range of lift coefficients but these differences are small and do not show any consistent trends (figs. 4 to 10 and reference 1).

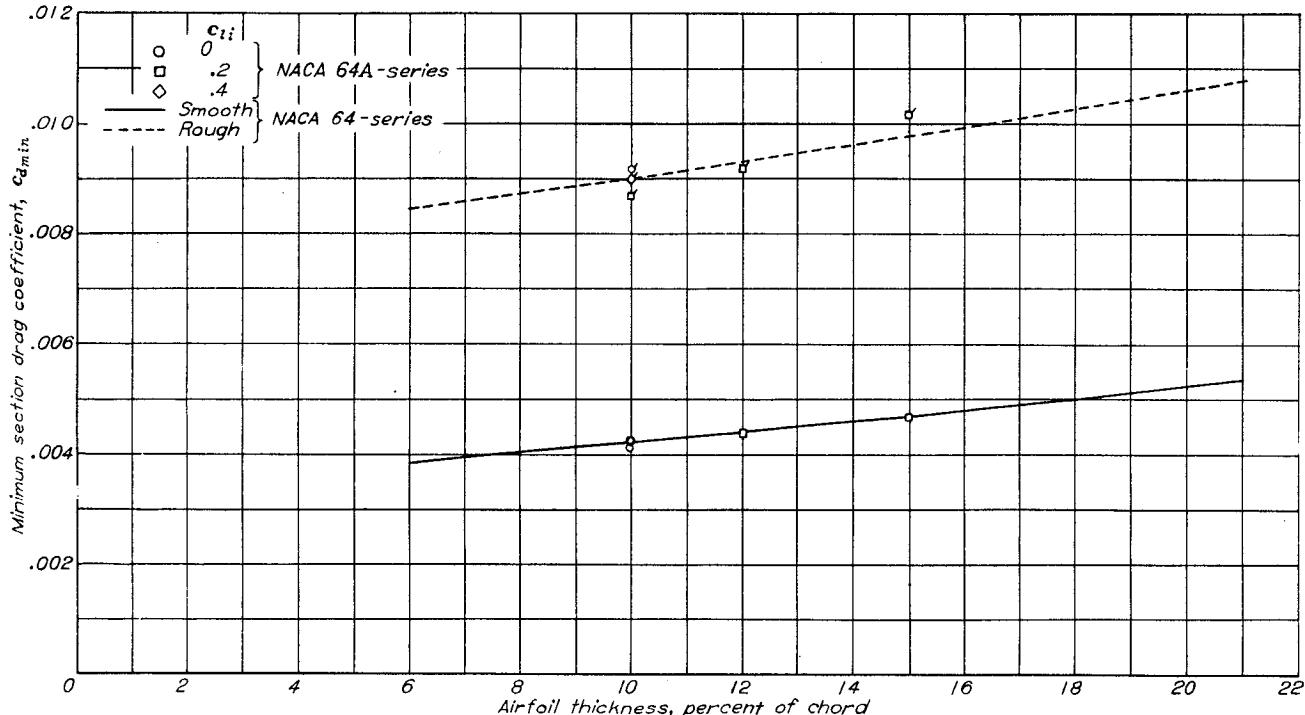


FIGURE 11.—Variation of minimum section drag coefficient with airfoil thickness for some NACA 64-series (reference 1) and NACA 64A-series airfoil sections of various cambers in the smooth condition and with standard leading-edge roughness.  $R=6 \times 10^6$ ; flagged symbols indicate NACA 64A-series sections with standard roughness.

**Lift.**—The section angle of zero lift as a function of thickness ratio is shown in figure 12 for NACA 64- and 64A-series airfoil sections of various cambers. These results show that the angle of zero lift is nearly independent of thickness and is primarily dependent upon the amount of camber for a particular type of mean line. Theoretical calculations made by use of the mean-line data of figure 3 and reference 1 indicate that airfoils with the  $a=0.8$  (modified) mean line should have angles of zero lift less negative than those with the  $a=1.0$  mean line. Actually, the reverse appears to be the case, and this effect is due mainly to the fact that airfoils having the  $a=1.0$  type of mean line have angles of zero lift which are only about 74 percent of their theoretical value (reference 1), and those having the  $a=0.8$  (modified) mean lines have angles of zero lift larger than indicated by theory.

The measured lift-curve slopes corresponding to the NACA 64-series and NACA 64A-series airfoils of various cambers are presented in figure 13 as a function of airfoil thickness ratio. No consistent variation of lift-curve slope with camber or Reynolds number is shown by either type of airfoil. The increase in trailing-edge angle which accompanies removal of the cusp would be expected to reduce the lift-curve slope by an amount which increases with airfoil thickness ratio (references 3 and 4). Because the present data for the NACA 6A-series sections show essentially no variation in lift-curve slope with thickness ratio, it appears that the effect of increasing the trailing-edge angle is about

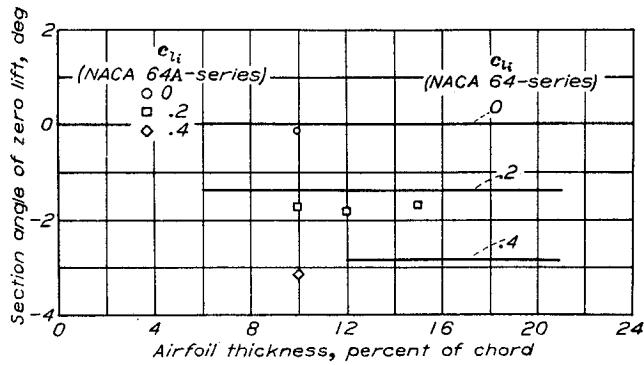


FIGURE 12.—Variation of section angle of zero lift with airfoil thickness ratio and camber for some NACA 64-series (reference 1) and NACA 64A-series airfoil sections.  $R=6\times 10^6$ .

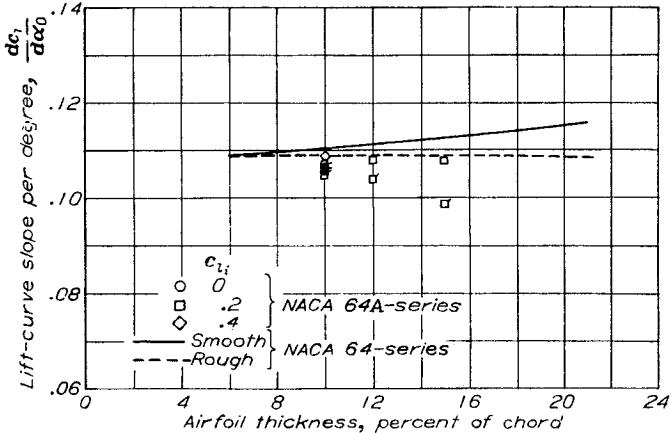
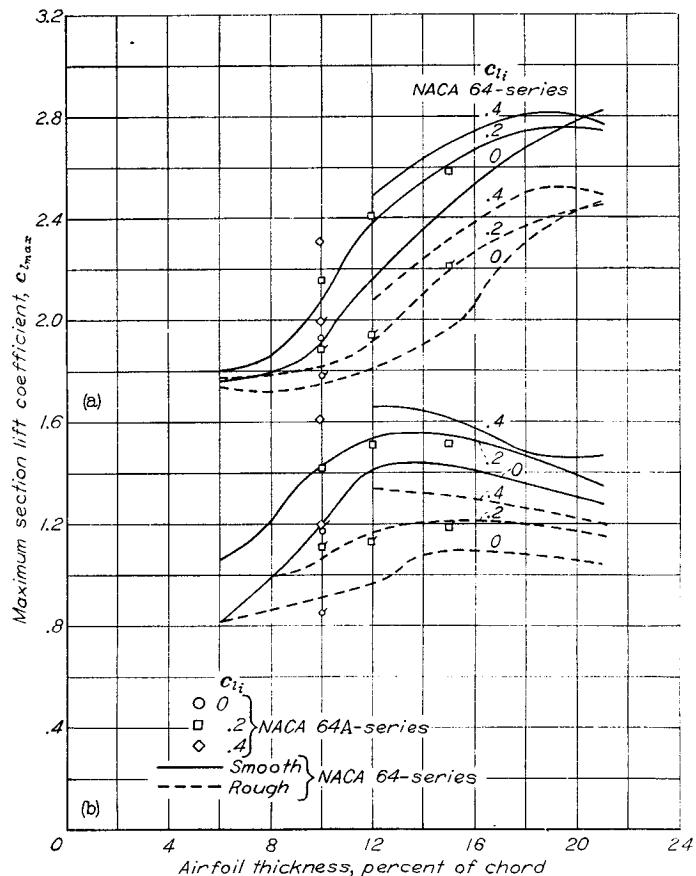


FIGURE 13.—Variation of lift-curve slope with airfoil thickness ratio for some NACA 64-series (reference 1) and NACA 64A-series airfoil sections of various cambers both in the smooth condition and with standard leading-edge roughness.  $R=6\times 10^6$ ; flagged symbols indicate NACA 64A-series sections with standard roughness.

balanced by the increase in lift-curve slope with thickness ratio shown by NACA 6-series sections. The value of the lift-curve slope for smooth NACA 64A-series airfoil sections is very close to that predicted from thin airfoil theory ( $2\pi$  per radian or 0.110 per degree). Removing the trailing-edge cusp from an airfoil section with standard leading-edge roughness causes the lift-curve slope to decrease quite rapidly with increasing airfoil thickness ratio.

The variation of the maximum section lift coefficient with airfoil thickness ratio and camber at a Reynolds number of  $6\times 10^6$  is shown in figure 14 for NACA 64-series and NACA 64A-series airfoil sections with and without standard leading-edge roughness and simulated split flaps deflected  $60^\circ$ . A comparison of these data indicates that the character of the variation of maximum lift coefficient with airfoil thickness ratio and camber is nearly the same for the NACA 64-series and NACA 64A-series airfoil sections. The magnitude of the maximum lift coefficient appears to be slightly less for the plain NACA 64A-series airfoils and slightly higher for the NACA 64A-series airfoils with split flaps than corresponding values for the NACA 64-series airfoils. These differences are small, however, and for engineering applications the maximum-lift characteristics of NACA 64-series and 64A-series airfoil sections of comparable thickness and design lift coefficient may be considered practically the same.



(a) Airfoil with simulated split flap deflected  $60^\circ$ .  
(b) Plain airfoil.

FIGURE 14.—Variation of maximum section lift coefficient with airfoil thickness ratio and camber for some NACA 64-series (reference 1) and NACA 64A-series airfoil sections with and without simulated split flaps and standard roughness.  $R=6\times 10^6$ ; flagged symbols indicate NACA 64A-series airfoils with standard roughness.

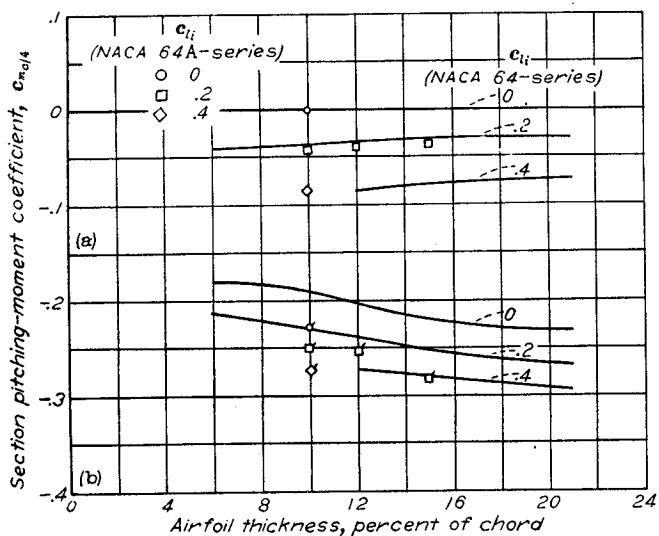
A comparison of the maximum-lift data for NACA 64A-series airfoil sections, presented in figures 4 to 10, with similar data for NACA 64-series airfoil sections indicates that the scale-effect characteristics of the two types of section are essentially the same for the range of Reynolds number from  $3 \times 10^6$  to  $9 \times 10^6$ .

**Pitching moment.**—Thin-airfoil theory provides a means for calculating the theoretical quarter-chord pitching-moment coefficients of airfoil sections having various amounts and

types of camber. Calculations were made according to these methods for airfoils having the  $a=1.0$  and  $a=0.8$  (modified mean lines by using the theoretical mean-line data presented in figure 3 and in reference 1. The results of these calculations indicate that the quarter-chord pitching-moment coefficients of the NACA 64A-series airfoil sections having the  $a=0.8$  (modified) mean line should be only about 87 percent of those for the NACA 64-series airfoil sections with the  $a=1.0$  mean line. The experimental relationship between the quarter-chord pitching-moment coefficient and airfoil thickness ratio and camber, shown in figure 15, discloses that the plain NACA 64A-series airfoils have pitching-moment coefficients which are slightly more negative than those for the plain NACA 64-series airfoils. The increase in the magnitude of the pitching-moment coefficient of NACA 64A series airfoils as compared with NACA 64-series airfoil becomes greater when the airfoils are equipped with simulated split flaps deflected  $60^\circ$ . A comparison of the theoretical and measured pitching-moment coefficients is shown in figure 16 for NACA 64-series and 64A-series airfoil sections. These comparative data indicate that the NACA 64A-series section much more nearly realize their theoretical moment coefficient than do the 64-series airfoil sections. Similar trends have been shown to result when mean lines such as the  $a=0.8$  type are employed with NACA 6-series airfoils (reference 1).

**Aerodynamic center.**—The position of the aerodynamic center and the variation of the moment coefficient with lift coefficient about this point were calculated from the quarter chord pitching-moment data for each of the seven airfoil tested. The variation of the chordwise position of the aerodynamic center with airfoil thickness ratio is shown in figure 17 for the NACA 64-series and 64A-series airfoil sections. Since the data for the NACA 64-series airfoils showed no consistent variation with camber, the results are represented by a single faired curve for all cambers. Following this same trend, the position of the aerodynamic center for the NACA 64A-series airfoils shows no consistent variation with camber. The data of figures 4 to 10 show that the variations in the Reynolds number have no consistent effect upon the chordwise position of the aerodynamic center.

Perfect fluid theory indicates that the position of the aerodynamic center should move rearward with increasing airfoil thickness and the experimental results for the NACA 64-series airfoil sections follow this trend. The data o



(a) Plain airfoil.  
(b) Airfoil with simulated split flap deflected  $60^\circ$ .

FIGURE 15.—Variation of section quarter-chord pitching-moment coefficient at zero angle of attack with airfoil thickness ratio and camber for some NACA 64-series (reference 1) and NACA 64A-series airfoil sections with and without split flaps.  $R=6 \times 10^6$ ; flagged symbols indicate NACA 64A-series airfoils with  $60^\circ$  simulated split flap.

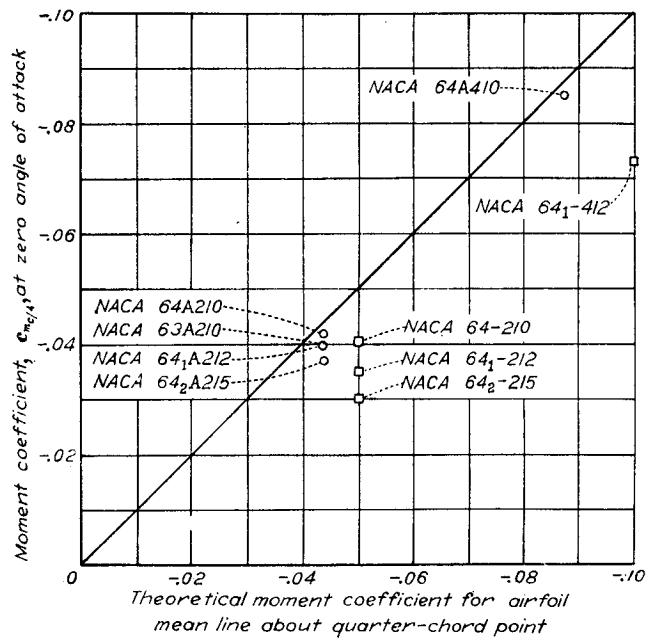


FIGURE 16.—Comparison of theoretical and measured pitching-moment coefficients for some NACA 64-series and 64A-series airfoil sections.  $R=6 \times 10^6$ .

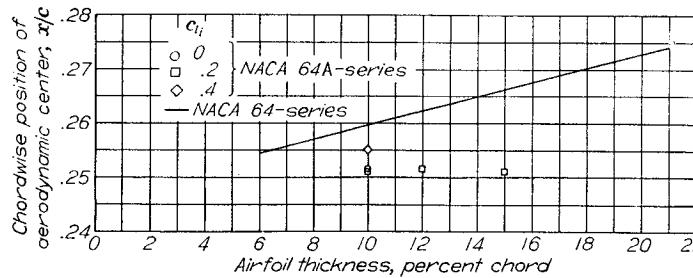


FIGURE 17.—Variation of chordwise position of aerodynamic center with airfoil thickness ratio for some NACA 64-series (reference 1) and 64A-series airfoil sections of different cambers.  $R=6 \times 10^6$ .

reference 5 show important forward movements of the aerodynamic center with increasing trailing-edge angle for a given airfoil thickness ratio. The results obtained for the NACA 24-, 44-, and 230-series airfoil sections (reference 1) reveal that the effect of increasing trailing-edge angle predominates over the effect of increasing thickness because the position of the aerodynamic center moves forward with increasing thickness ratio for these airfoil sections. For the NACA 64A-series airfoils (fig. 17) the aerodynamic center is slightly behind the quarter-chord point and does not appear to vary with increasing thickness. These results suggest that the effect of increasing thickness is counterbalanced by increasing trailing-edge angle for these airfoil sections.

### CONCLUSIONS

From a two dimensional wind-tunnel investigation of the aerodynamic characteristics of five NACA 64A-series and two NACA 63A-series airfoil sections the following conclusions based upon data obtained at Reynolds numbers of  $3 \times 10^6$ ,  $6 \times 10^6$ , and  $9 \times 10^6$  may be drawn:

1. The section minimum drag and maximum lift coefficients of corresponding NACA 6-series and 6A-series airfoil sections are essentially the same.
2. The lift-curve slopes of smooth NACA 6A-series airfoil sections appear to be essentially independent of airfoil thickness ratio, in contrast to the trends shown by NACA 6-series airfoil sections. The addition of standard leading-edge roughness causes the lift-curve slope to decrease with increasing airfoil thickness ratio for NACA 6A-series airfoil sections.

3. The section angles of zero lift of NACA 6A-series airfoil sections are slightly more negative than those of comparable NACA 6-series airfoil sections.

4. The section quarter-chord pitching-moment coefficient of NACA 6A-series airfoil sections are slightly more negative than those of comparable NACA 6-series airfoil sections. The position of the aerodynamic center is essentially independent of airfoil thickness ratio for NACA 6A-series airfoil sections.

LANGLEY MEMORIAL AERONAUTICAL LABORATORY,  
NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,  
LANGLEY FIELD, VA., May 6, 1947.

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